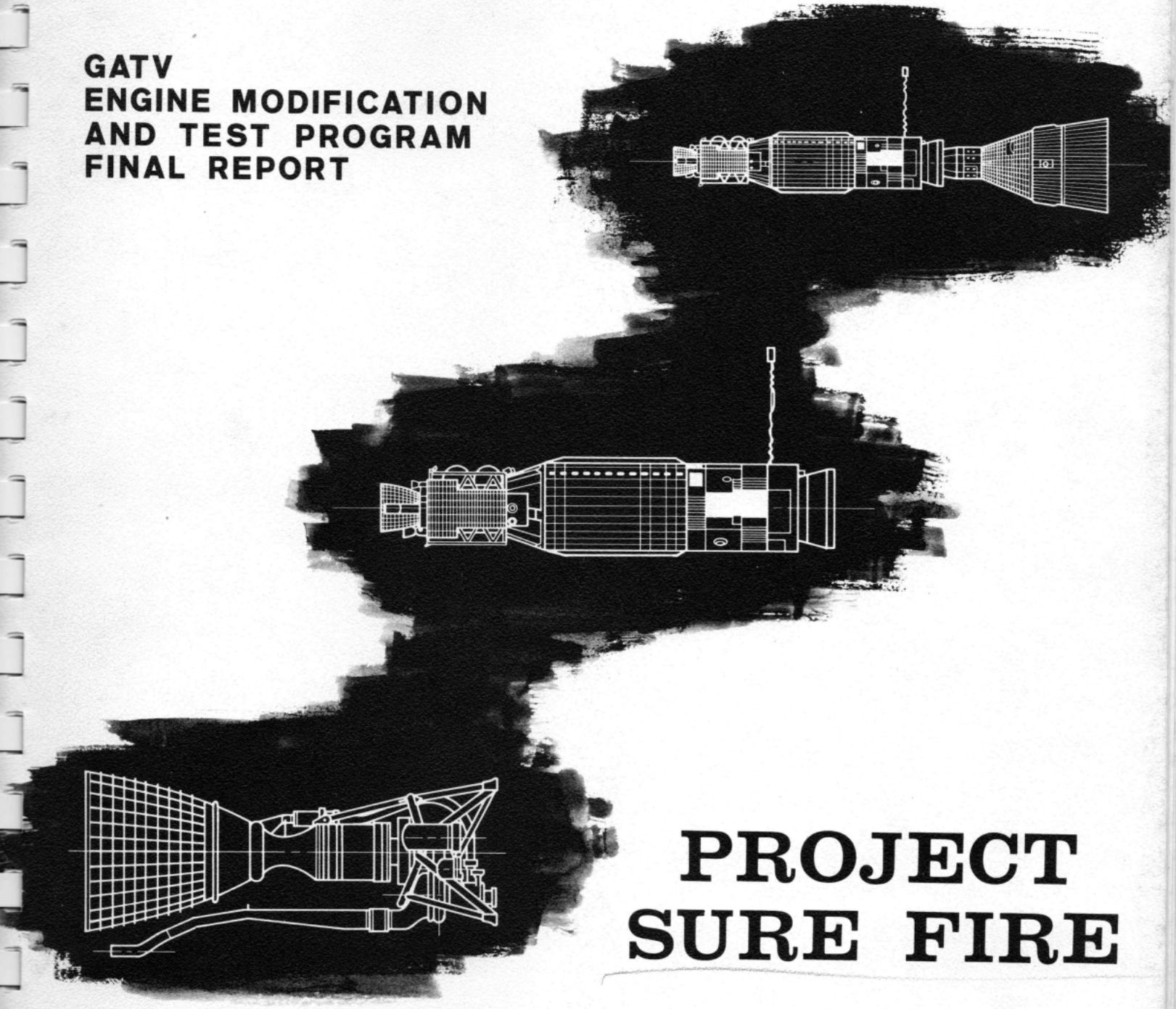
GATV ENGINE MODIFICATION AND TEST PROGRAM FINAL REPORT



PROJECT SURE FIRE

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VOLUME I

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FOREWORD

This document constitutes the final report of the Gemini Agena Target Vehicle Engine Modification and Test Program (Project Sure Fire). Project Sure Fire was concerned with the design modification, redevelopment, and flightworthiness demonstrations of the GATV multistart rocket engine. The design modification and test requirements of this program were established by and conducted in accordance with Engineering Change Proposal LH-545-101F-R2 to Contract AF 04(695)-545. LMSC Project Sure Fire activity was conducted in compliance with Air Force letter contract designated S. A. No. 25 to the AF 04(695)-545 contract. This report complies with the Contractor Data Requirements List (DD form 1423) line item No. 61, Final Test Report.

SUMMARY

The original USAF Model XLR 81-BA-13 Rocket Engine was designed and developed to provide a multistart primary propulsion system capability for the Gemini Agena Target Vehicle (GATV). The engine configuration was similar to the basic Standard Agena Model YLR 81-BA-11 Engine except for the start system, the main propellant valve sequencing which changed due to removal of pressure switches from the engine control circuit, and associated electrical system modifications. The main propellant valve sequencing provided a fuel lead ignition sequence in the main thrust chamber under normal operating conditions, creating a difference from the oxidizer lead designed into the YLR 81-BA-11 engine. Extensive sea level subassembly and engine testing and simulated altitude testing at a maximum of 120,000 feet were accomplished during development and Preliminary Flight Rating Tests (PFRT).

During the first flight of the multistart engine on GATV 5002 on 25 October 1965, premature engine shutdown occurred during the first engine start sequence. Engine shutdown caused propellant flow from the main tanks to stop, but the pressurization system continued to pressurize the small ullage volumes, eventually rupturing the tanks. Tank rupture allowed propellants to mix and react, destructing the vehicle. The postflight failure analysis resulted in the conclusion that the most probable cause of the flight failure was hard start.* Less probable causes of the failure were pyrotechnic separation shock damage, and inadvertent engine shutdown by the engine electronic gate.

Corrective action requirements were generated based on results of the postflight analysis, a propulsion system and vehicle aft rack design review, and a symposium on ignition of hypergolic propellants. The engine design change recommendations were to convert the XLR 81-BA-13 engine to a thrust chamber oxidizer lead start sequence

*In the main combustion chamber of the rocket engine a hard start is not a detonation, but a result of a reaction occurring at a very high rate.

similar to the YLR 81-BA-11 engine, incorporate shock mounting for certain engine electrical control components, and disable the electronic gate shutdown capability during ascent burn operation. Test requirements were established to verify adequacy of the design changes and demonstrate flightworthiness of the modified engine configuration. Results of a symposium on hypergolic ignition indicated that one significant test requirement was not included in the original XLR 81-BA-13 engine development and PFRT programs. This was engine testing at an altitude which properly simulated the hard vacuum that exists during flight. Therefore, an engine modification and test program was planned which required reliable ignition demonstration during hard vacuum simulation tests at 250,000 ft before the target launch date for GATV 5003. This required an accelerated, maximum success schedule. Consequently an Air Force, Aerospace, NASA, and industry team effort and maximum priorities were necessary to accomplish the program according to the proposed schedule. Project Sure Fire was established to manage the engine modification and test program, and activity was initiated early in November 1965.

Testing was immediately initiated on the turbine pump assembly. These tests provided the preliminary engine transient performance values, defined the initial detailed design change requirements, verified satisfactory operating characteristics of the proposed modified configuration prior to initiating engine level testing, and verified expected operating characteristics with various imposed malfunction conditions. The turbine pump testing included OFPS durability tests and other tests were conducted as necessary to verify design changes incorporated as the program progressed. A total of 75 turbine pump assembly tests were accomplished between November 1965 and March 1966.

A total of 37 gas generator/start system tests were conducted from November 1965 through March 1966. During these tests, conducted at sea level and at 240,000 feet simulated altitude, reliable gas generator ignition was achieved throughout the range of predicted flight operating conditions, as well as for conditions normally considered conducive to producing adverse ignition characteristics. In addition, reliable ignitions were demonstrated after a gas generator/start system had simulated a 28-day pad hold period and a subsequent 5-day altitude coast storage period.

A pressure switch relay box was designed for the initially proposed configuration, and the development and flightworthiness demonstration tests were conducted on this component in December 1965 and January 1966. Vibration, shock, humidity, acceleration, altitude, and electrical tests were conducted. A relay failure occurred during development vibration tests, and after a subsequent reliability analysis, the relay was removed and the relay box was converted to a junction box. Development and flightworthiness demonstration tests were continued on the relay box however, and flightworthiness of this component was demonstrated. The final configuration junction box was qualified by similarity. These tests also resulted in a design change of the shock mounting bonding straps, and provided the basis for improved relay manufacturing and quality control procedures.

Elevated stress tests, including vibration, shock, voltage, temperature, and extended power-on time tests were conducted, as applicable, on the electronic gate, pressure switch relay box, and engine electrical harness. Adequate design margins relative to the flight operating environment were demonstrated for these components.

The proposed engine modification involved addition of two pressure switches in the engine control circuit to provide the required thrust chamber oxidizer lead start sequence. Turbine pump assembly test results indicated a high frequency actuation-deactuation cycling characteristic of the backup oxidizer feed pressure switch during a normal engine start sequence. Pressure switch durability tests were conducted which verified that the mechanical durability of the feed pressure switch was adequate for the expected cycling characteristics during a flight mission. Pressure switch vacuum tests were conducted, with no degradation of the microswitch contacts observed, successfully demonstrating switch operational capability at the Gemini mission altitude for a minimum 5-day period.

Vibration, shock, and hot fire tests were conducted in January and February 1966 as part of the engine sea level flightworthiness demonstration (FWD) program. Satisfactory structural design of the new and modified component installations was verified. The 42 hot fire tests demonstrated satisfactory operation and sequencing of the modified engine configuration, and verified successful implementation and checkout of

the modified engine test and servicing procedures. The sea level FWD test program provided high confidence that the modified engine configuration provided the desired transient operating and sequencing characteristics over the range of specification temperatures, pressures, and voltages. Consequently, the simulated altitude FWD tests were allowed to proceed without further design change and with the objective of demonstrating that these transient characteristics would provide safe and reliable ignition and operation in a flight environment.

A total of 43 engine FWD tests at simulated altitudes ranging from 257,000 to 453,000 feet, and two checkout firings at 85,000 feet were conducted during the period of 7 February to 2 April 1966. The ignition confidence, simulated mission, low temperature, and malfunction tests at an average simulated altitude of 356,000 feet successfully demonstrated the high altitude flightworthiness of the modified XLR 81-BA-13 engine. Sufficient confidence in the reliability of engine ignition had been gained from the 27 Phase I and Phase II altitude tests completed by 4 March 1966 to assure GATV 5003 flightworthiness and allow commitment of the modified engine design to flight. Significantly, the postulated GATV 5002 flight failure mode was confirmed during the altitude malfunction tests, which showed that a fuel lead on the XLR 81-BA-13 engine would produce hard starts when tested at the proper altitude and that a reasonably high probability of hardware damage exists. Reevaluation of Vehicle 5002 data indicated that the engine damage incurred during the flight was similar to that observed during the last fuel lead test. In addition to the successful flightworthiness demonstration of the modified engine, the altitude tests provided data on altitude ignition characteristics over a temperature range from $+100^{\circ}$F to below zero.

An unexpected destructive hard start occurred during a checkout firing early in the altitude test program. Post test data analysis and testing showed that excessive water and alcohol contamination (approximately 85 percent) was introduced into the engine fuel system during the prefire propellant loading operation. The fuel system became contaminated with water during test cell down time for instrumentation and hardware repair. An abbreviated isopropyl alcohol flush procedure was conducted to remove water from the engine; however, the water and alcohol were not completely removed from the facility fuel system, resulting in entry of the contaminated fuel load into the engine.

Full-scale and sub-scale thrust chamber ignition tests were instituted at BAC to evaluate the effects of fuel contamination. Results showed that significant increases in ignition delay and peak pressures occur as the quantities of alcohol and water in the fuel are increased. These results, in addition to results of further analysis and tests at AEDC and LMSC, clearly supported the conclusion that the checkout test failure was caused by contaminated fuel.

Further ignition tests were conducted at BAC to investigate thrust chamber ignition characteristics with fuel, oxidizer, and simultaneous propellant leads over a range of operating temperatures and altitudes (ambient pressures). Considerable data was relatable to the XLR 81-BA-13 engine thrust chamber, and was usable as an aid in explaining the differences in ignition characteristics in the main thrust chamber with fuel and oxidizer leads.

At the same test conditions, the XLR 81-BA-13 engine thrust chamber produced significantly different ignition characteristics for a fuel lead start sequence compared to an oxidizer lead. Therefore, a comparative evaluation of the differences in ignition characteristics was made based on test data for the full-scale (engine) thrust chamber, the sub-scale thruster, and the engine gas generator assembly. The hardware design factors (such as feed configuration, injector type, residence time, etc.) which can affect ignition were reviewed, and the dependent conditions existing in the chamber at ignition, i.e., mixture ratio, density, ignition delay and ignition chemistry, were recorded or derived as the test variables of altitude, temperature and propellant lead were changed. The requirements for a reliable ignition were considered. The proper pressure and temperature must be generated in the fuel-oxidizer mixture during the induction period just prior to ignition, and a sufficient amount of oxidizer must be present during induction to prevent long ignition delays or quenching of the reaction.

Based on analysis of the design factors and conditions in the full-scale and sub-scale thrust chambers at ignition, and on data from the literature, it appears that the chemistry of the ignition is involved in producing the hard start experienced in the main thrust chamber with the fuel lead start sequence. When oxidizer is not present in sufficient quantities during the induction period, a suitable oxidation reaction

does not occur to overcome the effects that the hard vacuum produces during the propellant preflow and/or mixing period. Thus, proper pressures and temperatures are not developed, and a long ignition delay results, during which secondary reactions probably occur, producing high energy intermediate compounds. A highly reactable mixture forms, including the UDMH fuel which itself posses monopropellant characteristics. The resultant mixture becomes the source of the additional energy which produces the hard start when ignition occurs. In the XLR 81-BA-13 thrust chamber, additional damage was incurred because the residence time was such that a reactable mixture accumulated downstream of the throat during the long ignition delay, causing the nozzle overpressure when ignition occurred.

Although the gas generator operates reliably with a fuel lead, this reliability is attributable to the relatively very large volume of the gas generator/turbine manifold assembly which readily accommodates the energy stored at ignition, and to a pre-ignition pressure rise which indicates that a preigniter probably exists, similar to the main thrust chamber oxidizer lead start sequence.

CONCLUSIONS

Significant conclusions derived from Project Sure Fire are:

- Flightworthiness of the modified XLR 81-BA-13 engine configuration was successfully demonstrated.
- An oxidizer lead start sequence is optimum for the XLR 81-BA-13 engine thrust chamber, providing low and acceptable ignition shock levels over the range of required operating conditions.
- Significant differences exist between oxidizer and fuel lead ignition characteristics in the XLR 81-BA-13 thrust chamber.
- The GATV 5002 flight failure analysis conclusion that an engine hard start occurred was proven correct, and the postulation that the engine hard start was due to a fuel lead start sequence was also correct.
- Fuel lead hard starts yield high probability of damage to the thrust chamber assembly. Reevaluation of data from Vehicle 5002 indicates than an oxidizer line break occurred in the same area as that observed during the last fuel lead test at Arnold Engineering Development Center (AEDC). No reactions or adverse pressures were detected in any of the thrust chamber manifold cavities during the fuel lead starts at AEDC. The hard start reactions occurred in the combustion chamber and divergent nozzle.
- The fuel lead hard start mechanism appears to involve the chemistry of the reaction during the induction period. Lack of an excess of oxidizer apparently prevents a satisfactory oxidation reaction from occurring relative to that for an oxidizer lead start sequence. A very long ignition delay occurs, allowing an accumulation of a reactable oxidizer-fuel mixture which probably contains high energy intermediate compounds formed during this delay.

- The XLR 81-BA-13 engine gas generator assembly provides reliable ignition with a fuel lead start sequence within the range of operating requirements. Low peak pressures and very slow pressure rise rates are always obtained. These characteristics appear to be due to the large volume of the gas generator assembly, to the low potential energy in the chamber at ignition, and perhaps most important, to a preignition pressure buildup which is most probably attributable to the existance of a preigniter oxidizer flow.
- Testing at the proper simulated altitude to determine engine ignition reliability is a necessary and an extremely important phase of space-flight engine development.
- Propellant triple point (phase) data provides a reliable guideline for defining the minimum altitude test requirements. Further studies on the relation of phase data, propellant injection and expansion dynamics at hard vacuum, and presence of excess fuel or oxidizer, are recommended in order to advance the state of the art.
- Existing ground test technology is more than sufficient to properly simulate required altitude conditions for medium-sized spacecraft rocket engines.
- Sea level and altitude sub-scale, and sea level full-scale ignition tests can be a valuable adjunct to full-scale altitude testing. However, full-scale altitude tests must be conducted as final proof that complete simulation of all factors affecting the ignition process for a specific configuration have been demonstrated.
- It is believed that Project Sure Fire has contributed to an advancement of the state of the art in hard vacuum simulation requirements and ignition technology for medium-sized space vehicle rocket engines.

Based on results obtained during Project Sure Fire, it is recommended that future space vehicle rocket engine development programs include full-scale testing at simulated altitude pressures well below the propellant triple point pressures. In addition, sub-scale thrust chamber ignition testing and selected laboratory and full-scale tests early in the development program are recommended. Valuable information for use in the detailed rocket engine design can thus be obtained.

To provide a necessary understanding of the mechanisms involved during ignition in the XLR 81-BA-13 engine gas generator and thrust chamber assemblies, the following types of tests are recommended:

- Cold flow injection tests of a full-scale gas generator manifold-injector unit to determine if a preigniter oxidizer flow or its equivalent exists.
- Propellant flow tests through initially evacuated thrust chamber cooling and manifold cavities to determine if a significant difference exists in the amount of heat transferred to the oxidizer or fuel preflow.
- Additional sub-scale tests at simulated hard vacuum with shock measurements (accelerometers) to determine the ignition shock characteristics over a wider range of propellant leads and particularly, to obtain the oxidizer lead threshold below which high ignition shocks occur.
- Laboratory tests to investigate the chemistry of the IRFNA-UDMH reaction under oxidizer and fuel lead conditions including determination of the reaction products formed during the induction period.

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NOMENCLATURE

AEDC	Arnold Engineering Development Center
AFSSD	Air Force Space System Division
ARO	Arnold Research Organization
ASA J-box	Aft safe/arm J-box
ATP	Acceptance test procedure
BAC	Bell Aerosystems Company
DCD	Design control drawing
DP-1	Drain port – location 1
ECP	Engineering Change Proposal
EMI	Electromagnetic interference
E&P	Electrical and pneumatic (test console)
ETR	Eastern Test Range
FGGSV	Fuel gas generator solenoid valve (same as GGFV)
FVAP	Fuel valve actuation pressure
FVIP	Fuel venturi inlet pressure
FWD	Flightworthiness demonstration
FWDT	Flightworthiness demonstration tests
GATV	Gemini Agena Target Vehicle
GG	Gas generator
GGFV	Gas generator fuel valve (same as FGGSV)
GGOV	Gas generator oxidizer valve (same as OGGSV)
HPB	High point bleed
IPA	Isopropyl alcohol
IRFNA	Inhibited red fuming nitric acid
L*	Combustion chamber characteristic length
LMSC	Lockheed Missiles & Space Company
MFV	Main fuel valve
NASA	National Aeronautics and Space Administration
OFP	Oxidizer feed pressure
OFPS	Oxidizer feed pressure switch

OGGSV	Oxidizer gas generator solenoid valve (same as GGOV)
OIP	Oxidizer injector pressure
OMP	Oxidizer manifold pressure
OMPS	Oxidizer manifold pressure switch
OVIP	Oxidizer venturi inlet pressure
P_c	Thrust chamber pressure
PFRT	Preliminary flight rating test
PIP	(An electrical signal identifying an event)
POSV	Pilot operated solenoid valve
pisa	Pounds per square inch absolute
psig	Pounds per square inch gage
PSD	Power spectral density
PSJB	Pressure switch junction box
PSRB	Pressure switch relay box
PTVA	Propulsion test vehicle assembly
SCTB	Santa Cruz Test Base
SSD	Space Systems Division
TC	Thrust chamber
TCPS	Thrust chamber pressure switch
TED	Turbine exhaust duct
Tes-1	Temperature of engine surface – location 1
TFI	Temperature of fuel in feedline near pump inlet
TF-1	Temperature of fuel – location 1
TFPI	Temperature of at fuel pump inlet
Tgg-1	Temperature of gas generator – location 1
TM	Telemetry
TMP	Turbine manifold pressure
TOI	Temperature of oxidizer in feedline near fuel pump inlet
TO-1	Temperature of oxidizer – location 1
TOPI	Temperature of oxidizer at pump inlet
TOS	Turbine overspeed
TP-1	Test port – location 1
UDMH	Unsymmetrical dimethylhydrazine

Section 1
INTRODUCTION

1.1 DESCRIPTION

The Gemini Agena Target Vehicle (GATV) Engine Modification and Test Program (Project Sure Fire) was established by joint LMSC-USAF-Aerospace-BAC action on 20 November 1965 following the GATV 5002 failure. The program was one for developing corrective improvements to, and demonstrating flightworthiness of the rocket engine to assure successful missions on vehicles 5003 and subsequent. LMSC was technical director for the Sure Fire activity which included subcontract effort by Bell Aerosystems Corporation (BAC) and testing conducted at the Arnold Engineering Development Center (AEDC), Tullahoma, Tennessee.

1.2 BACKGROUND

Project Sure Fire was established subsequent to the engine flight failure of GATV 5002 on 25 October 1965. Immediately following the GATV 5002 failure, LMSC conducted an accelerated data reduction and analysis program to determine the cause(s). During early November 1965, flight failure reviews were presented to the Flight Safety Review Panel, and a symposium on hypergolic rocket engine ignition at altitude was held at LMSC. The symposium was attended by 16 leading propulsion scientists and engineers representing Aerojet General, Bell Aerosystems Company, Aerospace, U. S. Air Force, Rocket Propulsion Laboratory, Marquardt, Rocketdyne, Princeton University, and LMSC.

The findings and conclusions of the early November activity were essentially (1) that the flight failure of Vehicle 5002 was due to a "hard start" of the Agena Primary Propulsion System rocket engine, (2) the "hard start" was probably caused by the fuel lead into the engine thrust chamber prior to ignition (the reliable-starting Standard Agena engine uses an oxidizer lead), (3) that an engine modification and test program be initiated, and (4) that altitude tests must be conducted at simulated altitudes above 250,000 feet.

On 15 November, the combined recommendations of the study and review groups were presented by LMSC and BAC to AFSSD, NASA, and Aerospace representatives. The final GATV Engine Modification and Test Program (Project Sure Fire) proposal was presented to NASA at Houston, Texas on 20 November 1965. The proposal was approved and LMSC started work immediately to meet a 24 March launch date for Gemini VIII and Vehicle 5003.

During the ensuing period, the redesign, modification, and redevelopment testing of the Agena rocket engine was performed as described and presented in this report.

1.3 OBJECTIVES

The basic objectives of Project Sure Fire were:

- Provide an improved engine design for the primary propulsion system, consisting of the XLR 81-BA-13 (8247-475200) engine, modified to approach as closely as possible the oxidizer preflow characteristics of the YLR 81-BA-11 (8096-475300) Standard Agena engine, but retaining the start system to preserve the multiple restart capability.
- Modify the engine control circuitry as required, to achieve the oxidizer preflow, and inhibit the turbine overspeed control during the ascent phase to increase the probability of achieving orbit.
- Demonstrate flightworthiness of modifications through an adequate test program.

1.4 PROGRAM SCOPE

Project Sure Fire engine modification and test program included:

- Design analysis
- Detailed engine and interface redesign
- Specification, drawing, test plan and procedure, and documentation revision and preparation
- Flight and test engine modification
- Development, evaluation, acceptance, elevated stress, and formal demonstration tests
- Production of two new engines

Additional design, hardware modification, analysis and evaluation testing, not originally planned, were incorporated as required, due to results of initial test and program activities. Modifications under Project Sure Fire were effective for Vehicles 5001, and 5003 and up.

The first vehicle scheduled to incorporate the Sure Fire modification was GATV 5003, with a 24 March launch date requirement. Engines modified were flight engines 806, 807, and 808, and flightworthiness test engines 805 and 803. Component and subassembly testing included:

- Pressure switch relay box
- Electronic gate
- Turbine pump assembly and control hardware
- Thrust chamber (full scale injector flow and ignition tests and subscale ignition tests)
- Start system (gas generator)
- Engine electrical harness
- Pressure switches
- Modified hardware installation dynamic tests (engine XRM-4)

1.5 PROGRAM IMPLEMENTATION

Program implementation began with the removal of engines from Vehicles 5001 and 5003. These engines were shipped to BAC for modification to the oxidizer lead configuration (two pressure switches and a relay box added), including change to use the YLR 81-BA-11 engine valves and shock mounting installation of the electronic gate and relay box.

The test program was planned to include engine component, subassembly, and complete engine testing. Component design verification and elevated stress tests were performed at BAC, as well as turbopump testing at sea level and altitude tests on the engine start system. A flightworthiness demonstration test program was also performed at sea level at BAC on a complete engine. Complete engine flightworthiness demonstration testing was conducted at altitude conditions at Arnold Center test facilities. New design for flight hardware ran concurrently with the development test programs.

The following tests were conducted which were not included in the original test program:

- Oxidizer manifold transducer pressure sensing line dynamic tests
- Engine low-temperature ignition shock tests at altitude
- Pressure switch durability tests
- Pressure switch vacuum tests
- Full-scale thrust chamber ignition tests with contaminated propellants at sea level
- Sub-scale ignition tests at sea level and altitude
- Thrust chamber injection flow tests

1.6 REPORT PARTICULARS

This final report covers the significant aspects of Project Sure Fire activity from November 1965 to 1 July 1966. It describes GATV engine design modification and development (Section 2) and the modified engine start and shutdown sequence (Section 3). Sections 4 through 9 describe the supporting tests and flightworthiness demonstrations conducted in conjunction with Project Sure Fire. Section 10 discusses a comparison of fuel and oxidizer leads. Reliability engineering analysis and engine retrofit kits are covered in Sections 11 and 12, respectively.

Project Sure Fire effort remaining at the time this report is submitted includes review and submission of the BAC and ARO final reports on the AEDC Altitude FWD tests, and minimal final data analysis and evaluation activity. Any significant items resulting from this remaining effort will be submitted as addenda or errata to this report, if required.

Section 2
ENGINE DESIGN AND MODIFICATION HISTORY

2.1 BASIC ENGINE MODIFICATIONS

Results of the GATV 5002 post flight data analysis, a Primary Propulsion System design review, and a combustion symposium provided the basic engine redesign concepts and modification requirements. The investigating groups concluded that engine failure was due to a hard start and that the hard start probably was caused by a fuel rather than an oxidizer lead into the thrust chamber prior to ignition.

The primary recommendation of the investigating groups was to convert the XLR 81-BA-13 engine thrust chamber propellant flow sequence from a fuel lead to an oxidizer lead configuration (basically a return to the configuration used in the Standard Agena YLR 81-BA-11 engine). To achieve the oxidizer lead, a change in the main fuel valve control was required. Two redundant pressure switches in the oxidizer flow path to the combustion chamber were determined to be the most reliable and expedient means for controlling the lock-in relay electrical control circuit for the engine fuel valve. This arrangement was similar to the configuration used for a time on the YLR 81-BA-9 (BAC Model 8096) engine. Also, the modification would be compatible with a development test effort consistent with accelerated schedule requirements of the Gemini Program.

An evaluation was oonducted to select the optimum engine main propellant valves for the modified engine. This evaluation was based on the four valve configurations available from the YLR 81-BA-11 and XLR 81-BA-13 engines. The analysis indicated a significant advantage in minimizing oxidizer post flow by using the XLR 81-BA-13 engine main oxidizer valve configuration. However, consistent with the philosophy to convert the engine to the YLR 81-BA-11 engine thrust chamber propellant flow

control system, the YLR 81-BA-11 engine (Standard Agena) main propellant valves were selected for use on the modified multi-start XLR 81-BA-13 engine.

Additional design modifications proposed for the modified engine included a vehicle electrical control modification. This modification would maintain electrical power on the engine electronic gate relay reset coils during ascent engine operation to disable the turbine overspeed shutdown capability during ascent. A shock mounting installation was incorporated for the engine electronic gate and the new pressure switch relay box installation to reduce the effects of engine operating dynamic environments and particularly, vehicle pyrotechnic shock effects on these components. For the proposed engine configuration, designated LMSC P/N 1461969-7, propellant flow and simplified electrical schematic diagrams are presented in Figures 2-1 and 2-2, respectively.

2.2 DESIGN CHANGES DURING REDEVELOPMENT

Changes from the basic redesign plan described in the preceding paragraph occurred as the redevelopment program progressed. The changes are discussed as follows.

2.2.1 Oxidizer Feed Pressure Switch

The initial turbine pump assembly development tests with the XLR 81-BA-13 engine start system indicated oxidizer pump start transient discharge pressure perturbations. Pressures were of sufficient magnitude to prematurely actuate the YLR 81-BA-9 engine configuration oxidizer feed pressure switch (OFPS) which was set to actuate at 235 psi. The duration of these pressure pulse OFPS actuations was sufficient to energize the pressure switch relay, which has electrical lock-in capability, and thus result in premature fuel valve actuation. Consequently, on 23 November 1965 the OFPS setting was changed to 391 psi, utilizing the same basic switch configuration. The higher OFPS actuation pressure satisfactorily eliminated the premature fuel valve actuation condition. This change, however, resulted in the OFPS providing a back-up function due to the higher turbine speed required to actuate the switch rather

than providing planned redundancy in both time and function. In addition, the oxidizer system pressure perturbations produced an OFPS actuation-deactuation chatter condition which required additional testing to verify safe and reliable pressure switch operation (durability) with the adverse cycle operation during the engine start and shutdown transients. After successful verification testing, the OFPS was determined to be satisfactory in terms of cyclic fatigue capability and in providing a reliable backup function to the oxidizer manifold pressure switch (OMPS).

2.2.2 Engine Electrical Circuit

Continuing design evaluation and review activity disclosed that a reduced electrical current capability rating of the engine electronic gate relays existed. The relay vendor indicated that the revised current rating was significantly below the design values required to supply electrical power through the relays to the two gas generator solenoid valves and the pilot operated solenoid valve. Consequently, on 3 December 1965 an electrical control circuit design change was initiated to provide electrical power to the gas generator solenoid valves, pressure switch relay box, and pilot operated solenoid valve from the vehicle aft safe arm junction box (ASA J-box) as shown in Figure 2-3. This modified electrical control circuit deleted the electronic gate from the series of components required to start and operate the rocket engine. Consequently, by deleting the electronic gate arm power, a positive disable of the turbine overspeed shutdown capability was achieved in lieu of the over-ride function originally proposed for the ascent burn. During subsequent engine operations on orbit, the arm power is applied to the electronic gate via the vehicle ASA J-box, allowing a shutdown signal to be sent to the vehicle control circuit in the event of potentially destructive turbine overspeed.

Additional electrical circuitry changes incorporated concurrently with the overspeed shutdown disable design change included (1) the addition of inductive spike suppression diodes in the vehicle ASA J-Box to suppress spikes from the engine solenoids, (2) addition of a post shutdown telemetry capability to monitor pressure switch position, (3) incorporation of a separate telemetry ground return for the pressure switch relay

box telltale, and (4) addition of the electronic gate switch group to vehicle telemetry.

The electronic gate design was not changed on the Sure Fire Program. However, additional quality control and inspection test requirements were incorporated and new (-7) electronic gates in accordance with these more stringent requirements were fabricated and tested for the flight engines. The -7 gate used on Vehicle 5003 performed satisfactorily on orbit, and the disable circuit functioned properly during ascent.

2.2.3 Pressure Switch Sensing Tube Assembly

The oxidizer feed pressure switch installation initially selected for the XLR 81-BA-13 engine was identical to the design previously used on the YLR 81-BA-9 engine. However, the YLR 81-BA-9 engine pressure switch pressure sensing tube assembly physically interfered with GATV aft rack hardware during thrust chamber maximum gimbal deflections and a tube assembly modification was required. A mockup modified tube assembly was fabricated at LMSC on 20 December 1965 and sent to BAC to define the OFPS sensing tube assembly design configuration required for the XLR 81-BA-13 engine. In addition, the oxidizer feed pressure switch installation required relocating the oxidizer manifold pressure transducer to a position on the engine oxidizer manifold line forward of the installation utilized on Vehicle 5002. This transducer installation modification reduced the length of the transducer pressure sensing line; however, the unsupported length of line remained in excess of standard design limits. Dynamic tests were conducted which verified that the tube assembly was capable of withstanding the flight dynamic environment and a redesign to decrease unsupported line length was considered unnecessary. This configuration performed satisfactorily during the nine burns of Vehicle 5003

2.2.4 Bonding Strap Redesign

The bonding straps for the pressure switch relay box shock mounting installation developed a fatigue failure during development dynamic tests of the relay box. Consequently, a design change of the bonding straps was initiated approximately

10 January 1966 for the pressure switch relay box and the electronic gate shock mounting installation. The redesigned straps satisfactorily passed dynamic requirements and demonstrated proper integrity during the formal demonstration tests.

2.3 FINAL ELECTRICAL CONTROL SYSTEM DESIGN (Effects of relay failure and reliability Analysis)

The initial engine electrical control system and associated vehicle Aft Safe Arm J-Box electrical circuit configuration was incorporated into the BAC turbine pump development test cell. An engine electrical control system design review and reliability analysis was performed in mid-January utilizing the results of the turbine pump development tests and relay box failure during development tests (relay failure is discussed later in the report). As discussed in paragraph 2.2.1, the turbine pump test data showed that an adverse pressure environment existed at the OFPS during a typical engine start transient with a potential OFPS microswitch spring fatigue failure. Consequently, additional tests were required to demonstrate the switch high-frequency cycle durability capability when subjected to this transient pressure environment.

The relay failure was attributed to contamination introduced during fabrication by the vendor, Allied Controls. Strict quality control provisions were instituted and new high-reliability relays were obtained for use on the program. Design, data, and reliability analysis results indicated the "OMPS, OFPS, and lock-in relay" configuration for the POSV electrical control provided excessive and potentially harmful redundancy. This configuration provided high reliability for applying and maintaining electrical power to the pilot operated solenoid valve (POSV), but failure mode analysis indicated that power could be applied prematurely to the POSV. Thus, there was reduced assurance that an oxidizer lead would be obtained and that a reliable start sequence would always result.

Consequently, considering the analysis results, a design change was proposed to eliminate the oxidizer feed pressure switch and utilize a high-reliability OMPS lock-in relay with POSV control circuitry similar to the YLR 81-BA-11 engine configuration.

The proposed electrical circuit design change was submitted to the Air Force during a design review meeting. Air Force direction was subsequently received on 21 January 1966 to redesign the pressure switch relay box to reduce the POSV electrical control circuit redundancy by eliminating the lock-in relay. Pending flightworthiness demonstration of the OMPS-OFPS electrical control circuit, shown in Figure 2-4, a parallel effort continued on (1) an OFPS disable kit, (2) production and acceptance test of two pressure switch relay boxes with the improved relay manufacturing and quality control corrective action requirements established subsequent to the development test failure, and (3) pressure switch relay box flightworthiness demonstration and elevated stress testing. The pressure switch junction box, shown in Figure 2-4, was developed and "qualified by similarity" to the pressure switch relay box. Consequently, the relay box flightworthiness demonstration and elevated stress testing served a dual purpose. The OMPS, OFPS, and pressure switch junction box electrical control circuit as shown in Figure 2-4 was the final design configuration which demonstrated successful operation during the subsequent test programs and flight of Vehicle 5003.

2.4 RETROFITS AND KIT FABRICATION

The shock mounting electrical bonding strap redesign, the OFPS pressure sensing tube assembly routing design change, and replacement of the pressure switch relay box with the pressure switch junction box resulted in the XLR 81-BA-13 engine configuration change from the LMSC P/N 1461969-7 (BAC P/N 8247-475200-11) to the LMSC P/N 1461969-9 (BAC P/N 8247-475200-13) engine configuration. These design changes for engines 808 and 806 installed on Vehicles 5003 and 5004, respectively, were incorporated by retrofit at the launch base. Engines 807, 809, and 810 were delivered (DD-250) in accordance with the LMSC P/N 1461969-9 configuration. The pressure switch junction box electrical system was incorporated for the safety limits and malfunction test series on the sea level flightworthiness demonstration test program, and was utilized during all engine altitude tests at AEDC.

The OFPS disable kit and an oxidizer injector pressure transducer installation kit were two engine design changes which were not incorporated into the final engine design. The Sure Fire program schedule did not allow sufficient time to delay oxidizer injector pressure kit design and fabrication until after the dynamic test results described in paragraph 2.2.3 were obtained. Therefore, in case the dynamic (vibration) tests indicated a requirement, a limited quantity of these kits was designed and fabricated to support the initial GATV flights. As indicated earlier, dynamic test results deleted the requirement for these kits.

2.5 FLIGHT CERTIFICATION PHILOSOPHY

Flightworthiness certification of the XLR-81-BA-13 engine was based on satisfactory completion of all tests relating to flightworthiness demonstration (FWD) as called out in Project Sure Fire Work Statement, LMSC ECP LH-545-101F R2. The flightworthiness test conditions and levels indicated in the work statement were selected as those which would provide adequate demonstration that the engine ignites and starts reliably and within specification limits during flight. Both LMSC and the Air Force concurred with the test conditions and levels specified in the work statement.

The POSV J-Box, P/N 8247-472383-7, which is now used on the XLR 81-BA-13 engine, was not subjected to component level flightworthiness demonstration tests, however, the pressure switch relay box, P/N 8247-472483-1 was successfully subjected to such tests. The POSV J-Box was judged to be flightworthy by virtue of its similarity to the pressure switch relay box. The two boxes are identical except as follows:

a. Allied relay, BAC P/N 8081-475026-1, is in the relay box but not in the POSV J-Box.
b. One MIL-SPEC resistor which is not in the relay box has been added to the POSV J-Box.

All flightworthiness tests for the XLR 81-BA-13 engine were conducted in accordance with LMSC quality assurance procedures (SSD Procedure Q-106) and BAC quality assurance procedures (QP 80.01A). The flightworthiness tests were witnessed on a

100 percent basis by BAC inspection and were monitored by LMSC Product Assurance and BAC Quality engineers. All test conditions were designed to meet or exceed Design Control Specification requirements (LMSC 1414463), and reflect the applicable engine environmental philosophy and test conditions required for the Gemini Agena Target Vehicle. The BAC failure reporting and evaluation system was utilized throughout all testing to assure adequate documentation of all suspected anomalies.

After review of all test conditions and data by LMSC and BAC personnel, the engine was determined to be flightworthy. An abbreviated formal certificate of flightworthiness was prepared, signed, and transmitted by LMSC Product Assurance prior to Vehicle 5003 flight.

Also, formal certifications by LMSC Sure Fire Engineering were submitted at the NASA Design Certification Review Meeting held prior to GATV 5003 flight.

2.6 PRODUCTION ENGINE MODIFICATIONS, TEST, AND CHANGE HISTORY

The tabulation below summarizes the original Sure Fire modification to each XLR 81-BA-13 engine, the configuration of the engine when acceptance tested, and the final configuration when flown or at the date of this report. Engines 803, 805, 806, 807, and 808 were returned to BAC and modified according to the Sure Fire work statement. Engines 809 and 810 are new engines which were contracted to be built in compliance with the Sure Fire work statement.

Table 2-1 shows the LMSC engine configuration dash numbers of each production flight engine as received at BAC prior to modification, the engine configurations as originally modified according to Sure Fire requirements, and the final configuration prior to flight. Table 2-2 delineates each of the modifications for each engine and the configuration when acceptance tested and flown.

Table 2-1

ENGINE CONFIGURATIONS

Engine S/N	Before Sure Fire Program	Initial Sure Fire Modification (LMSC 1411969)	Final Engine Configuration (LMSC 1461969)
806	LMSC P/N 1461969-5) (BAC P/N 8247-475200-7)	-7*	-9**
807	Originally GFE, no LMSC part number, BAC original P/N 8247-475200-9	-9	-9
808	Originally GFE, no LMSC part number, BAC original P/N 8247-475200-9	-7	-9
809	New engine	-9	-9
810	New engine	-9	-9

*The -7 engine changes are indicated in the Modifications Summary under Initial Sure Fire Modifications for Engine 808 and include the Electronic Gate P/N 8247-472283-7 which replaced gate P/N 8247-472283-5.

**The -9 engines differ from the -7 engine in that:

(a) POSV Relay Box P/N 8247-472-483-1 was replaced with POSV J-Box P/N 8247-472383-7.

(b) The pressure sensing tubing from the oxidizer valve to the oxidizer feed pressure switch was modified to conform to vehicle envelope requirements.

(c) Redesigned bonding straps for the pressure switch J-Box and electronic gate shock mounting installation were incorporated on the 1461969-9 engine configuration.

Table 2-2

PRODUCTION ENGINE MODIFICATION, TEST, AND CHANGE SUMMARY

Engine 808, BAC P/N 8247-475200-9, No LMSC P/N

Initial Sure Fire Modifications	Acceptance Test Configuration	Final Flight or FWD Test Configuration
MODIFIED TO LMSC 1461969-7 Added ox manifold pressure switch. Added ox feed press. sw. Added POSV relay box. New Engine Cable Assy. P/N 8247-475481-1. scheduled but not added. Added tubing and connect ox feed press sw. Electronic gate -7 scheduled but not added. Added shock mounts on relay box and electronic gate. Added bonding straps to relay box and electronic gate. Replaced 8247 main fuel valve with 8096 main fuel valve. Replaced 8247 main ox valve with 8096 main ox valve.	SAME AS AT DD-250 EXCEPT: Test cable used instead of cable assy. 8247-475481-1 (See Note 1.) Acceptance test conducted with elect. gate -5.	MODIFIED TO LMSC 1461969-9 POSV relay box P/N 8247-472483-1 replaced with POSV J-Box P/N 8247-472383-7. Cable assy. P/N 8247-475481-1 installed. Tubing to ox feed press. switch modified for clearance purposes Elect. Gate -7 installed on engine Changed bonding straps on POSV J-Box and elect. gate.
Engine 805, BAC P/N 8247-475200-7, LMSC P/N 1461969-5		
MODIFIED TO LMSC 1461969-7 Same changes as for engine 808. New engine cable assy. P/N 8247-475481-1 scheduled but not installed. Elect. gate -7 scheduled but not installed.	SAME AS AT DD-250 EXCEPT: Test cable used instead of cable assy. 8247-475481-1. Acceptance test conducted with elect. gate -5.	AEDC TEST CONFIGURATION POSV relay box P/N 8247-472483-1 replaced with POSV J-Box P/N 8247-472383-1. (See Note 2.) Cable assy. P/N 8247-475481-1 installed. Later modified to allow OMPS and OFPS monitoring simultaneously. Tests performed using elect. gate -5.

Table 2-2 (Cont.)

Engine 803, BAC P/N 8247-475200-7, LMSC P/N 1461969-5		
Initial Sure Fire Modifications	Acceptance Test Configuration	Final Flight or FWD Test Configuration
MODIFIED TO LMSC 1461969-7 Same changes as for engine 808. New engine cable assy. P/N 8247-475481-1 scheduled but not installed. Elect. gate -7 scheduled but not installed.	SAME AS AT DD-250 EXCEPT: Test cable used instead of cable assy. 8247-475481-1. (See Note 1.) Prior to acceptance test on 23 December 1965. Acceptance test conducted with elect. gate -5.	BAC AND AEDC TEST CONFIGURATION POSV relay box P/N 8247-472481-1 installed for part of BAC FWD tests. POSV J-Box P/N 8247-472383-7 installed for remainder of BAC FWD tests and AEDC tests. Cable assy. P/N 8247-475481-1 installed. Tests performed using elect. gate -5. Thrust Chamber 8247-470001-3 replaced with Std. Agena Thrust Chamber, 8096-470001-21, S/N 567 for AEDC tests.
Engine 806, BAC P/N 8247-475200-7, LMSC P/N 1461969-5		
MODIFIED TO LMSC 1461969-7 Same changes as for engine 808. New engine cable assy. P/N 8247-475481-1 scheduled but not installed. Elect. gate -7 scheduled but not installed.	SAME AS AT DD-250 EXCEPT: Test cable used instead of cable assy. 8247-475481. Acceptance test on 30 December 1965 Acceptance test conducted with electronic gate -5	MODIFIED TO LMSC 1461969-9 Same changes as for engine 808.

Table 2-2 (Cont.)

Engine 807, BAC P/N 8247-475200-9, No LMSC P/N		
Initial Sure Fire Modifications	Acceptance Test Configuration	Final Flight or FWD Test Configuration
MODIFIED TO LMSC 1461969-9 Added ox manifold press. switch. Added ox feed press. sw. Added POSV J-Box P/N 8247-472383-7. Added new cable assy. P/N 8247-475481-1. Pressure sensing tubing to ox feed press. sw. modified to fit within required engine envelope. Elect. gate -7 scheduled but not installed. Added shock mounts for POSV J-Box and Elect. Gate. Added modified bonding straps on POSV J-Box and elect. gate Replaced 8247 main fuel valve with 8096 main fuel valve. Replaced 8247 main ox valve with 8096 main ox valve.	SAME AS AT DD-250 EXCEPT: Acceptance test conducted with electronic gate -7 (S/N 24). After acceptance tests gate removed from engine 807 for installation on engine 808. Engine 807 was shipped without a -7 gate.	Elec. gate -7 installed on engine.
Engine 809, New Engine		
MANUFACTURED AS LMSC 1461969-9	SAME AS AT DD-250 (P/N 1461969-9)	EXPECTED TO BE SAME AS DD-250.

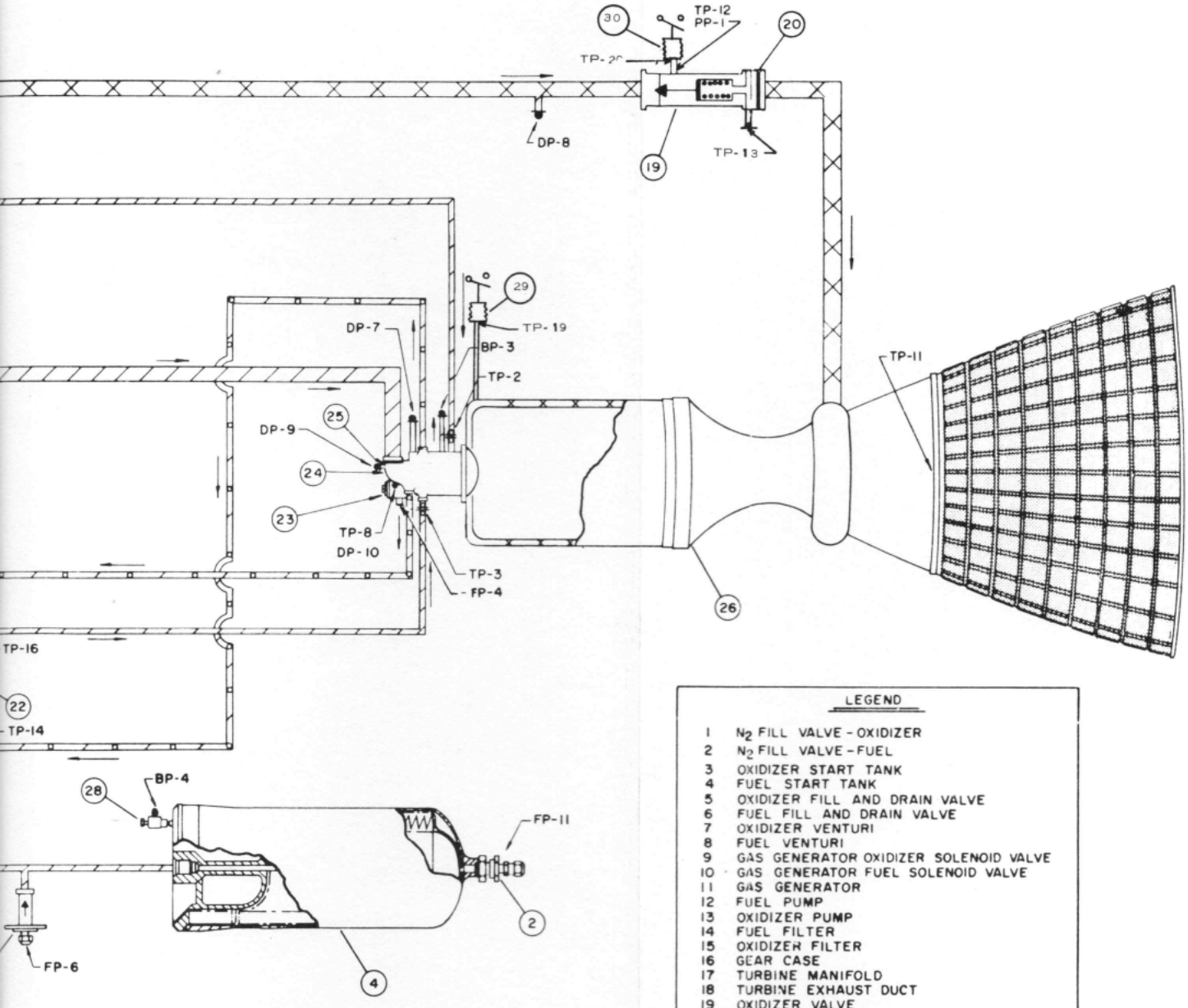
Table 2-2 (Cont.)

Engine 810, New Engine		
Initial Sure Fire Modifications	Acceptance Test Configuration	Final Flight or Fwd Test Configuration
Manufactured as LMSC 1461969-9	Same as DD-250 (P/N 1461969-9)	EXPECTED TO BE SAME AS AT DD-250.

NOTES:

(1) Acceptance tested with P/N 8247-475291-1 engine electrical harness (test cable). Test configuration provided electrical power to GG solenoid valves and POSV, including relay box through electronic gate in lieu of direct power from simulated vehicle aft safe arm J-Box. Test cable required pending test cell modification to incorporate simulated ASA J-Box.

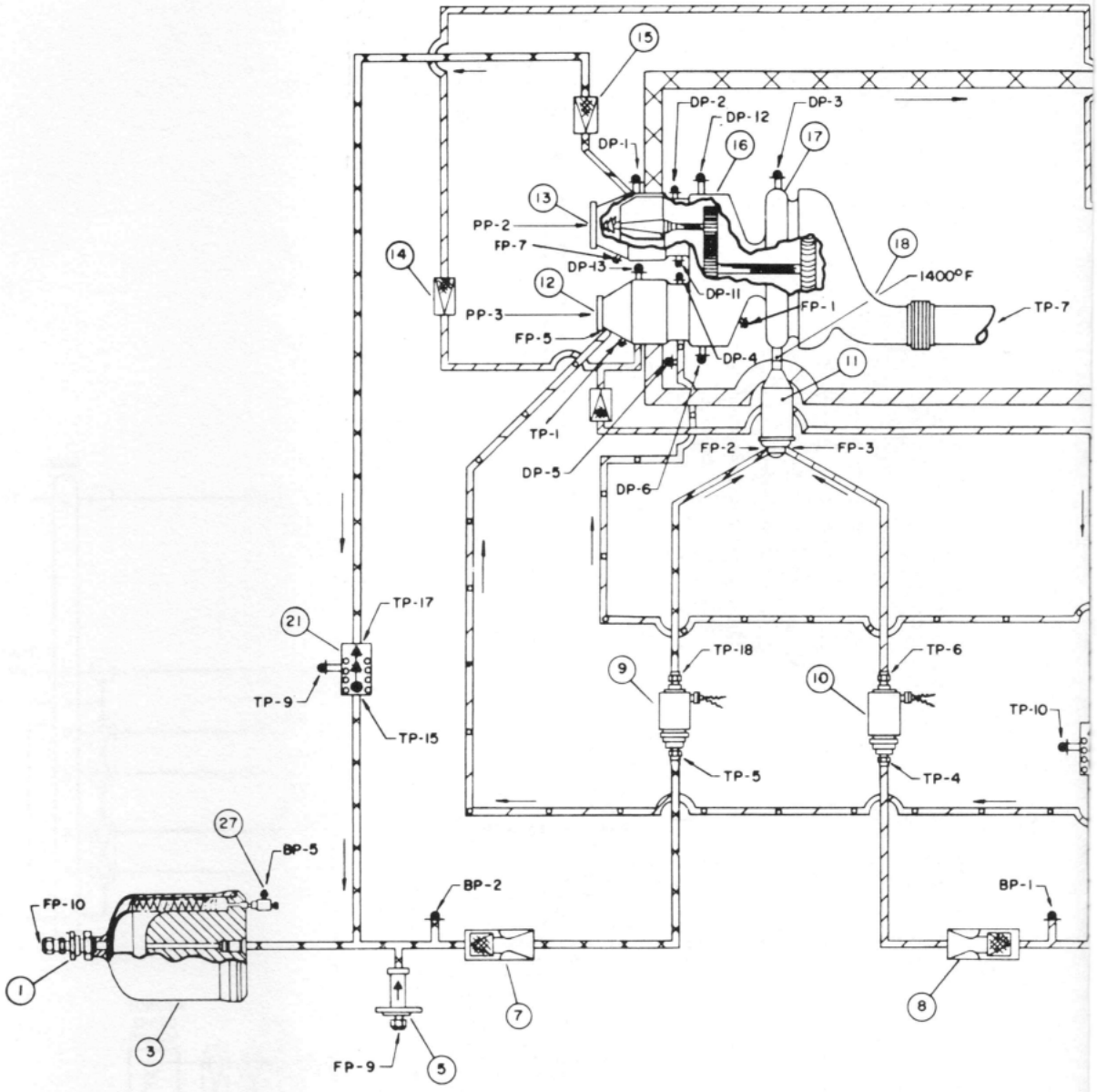
(2) Junction box installed at AEDC with additional engine configuration changes required for test (reference paragraph 7.4).



LEGEND

1	N_2 FILL VALVE - OXIDIZER
2	N_2 FILL VALVE - FUEL
3	OXIDIZER START TANK
4	FUEL START TANK
5	OXIDIZER FILL AND DRAIN VALVE
6	FUEL FILL AND DRAIN VALVE
7	OXIDIZER VENTURI
8	FUEL VENTURI
9	GAS GENERATOR OXIDIZER SOLENOID VALVE
10	GAS GENERATOR FUEL SOLENOID VALVE
11	GAS GENERATOR
12	FUEL PUMP
13	OXIDIZER PUMP
14	FUEL FILTER
15	OXIDIZER FILTER
16	GEAR CASE
17	TURBINE MANIFOLD
18	TURBINE EXHAUST DUCT
19	OXIDIZER VALVE
20	OXIDIZER FRANGIBLE DISC (RUPTURE - 180 PSI)
21	CHECK VALVE AND BLEED PORT (OXIDIZER)
22	CHECK VALVE AND BLEED PORT (FUEL)
23	PILOT-OPERATED SOLENOID VALVE
24	FUEL VALVE
25	FUEL FRANGIBLE DISC (RUPTURE - 525 PSI)
26	THRUST CHAMBER
27	OXIDIZER BLEED VALVE
28	FUEL BLEED VALVE
29	OXIDIZER MANIFOLD PRESSURE SWITCH
30	OXIDIZER FEED PRESSURE SWITCH
	OXIDIZER
	FUEL RETURN
	ORIFICE AND FILTER
	DRAIN, BLEED OR TEST POINT
	FUEL

Fig. 2-1 Modfied Engine Propellant Flow Diagram


15
DP-2
DP-12
DP-3
DP-1
16
17
13
PP-2
FP-7
DP-13
14
12
PP-3
FP-5
DP-11
FP-1
18
1400°F
TP-7
DP-4
11
TP-1
DP-5
DP-6
FP-2
FP-3
21
TP-17
TP-9
TP-15
9
TP-18
10
TP-6
TP-10
TP-5
TP-4
27
BP-5
FP-10
BP-2
BP-1
1
3
7
8
FP-9
5

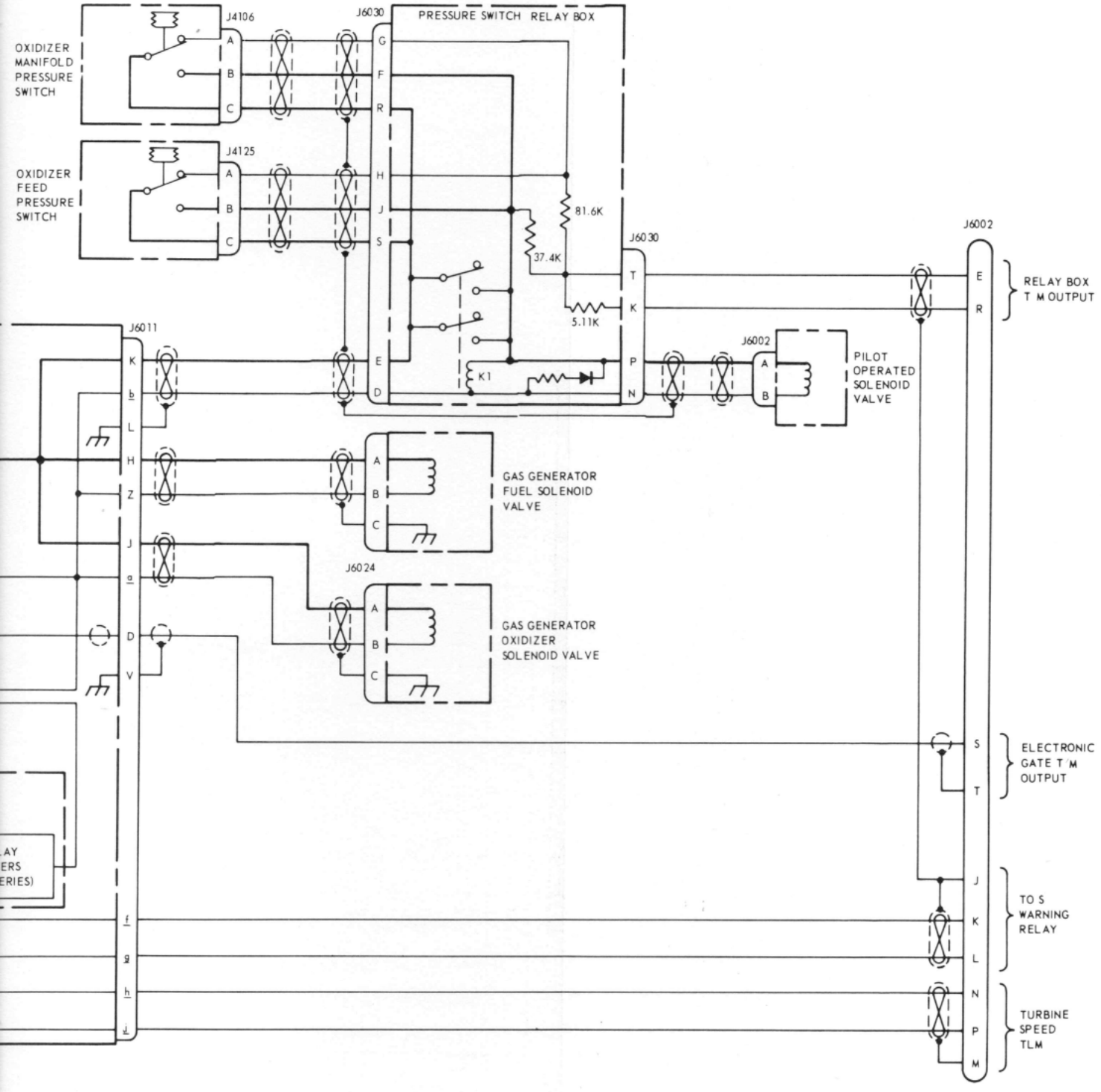
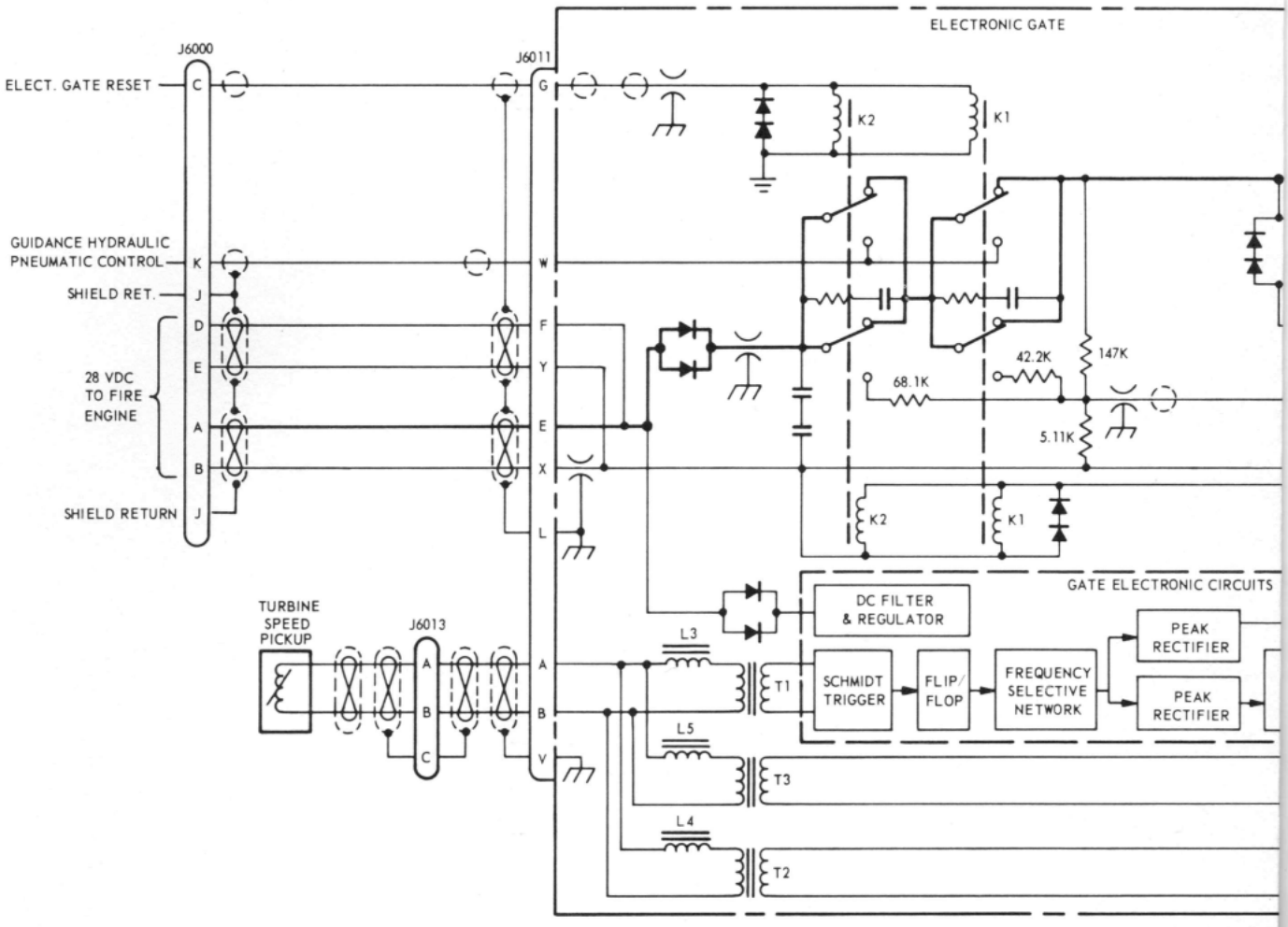


Fig. 2-2 Proposed Engine Electrical Control Circuit


ELECTRONIC GATE
J6000
J6011
ELECT. GATE RESET
C
G
GUIDANCE HYDRAULIC
PNEUMATIC CONTROL
K
W
SHIELD RET.
J
D
F
E
Y
28 VDC
TO FIRE
ENGINE
A
E
B
X
SHIELD RETURN
J
L
K2
K1
42.2K
147K
68.1K
5.11K
GATE ELECTRONIC CIRCUITS
DC FILTER
& REGULATOR
TURBINE
SPEED
PICKUP
J6013
L3
T1
SCHMIDT
TRIGGER
FLIP/
FLOP
FREQUENCY
SELECTIVE
NETWORK
PEAK
RECTIFIER
PEAK
RECTIFIER
L5
T3
L4
T2

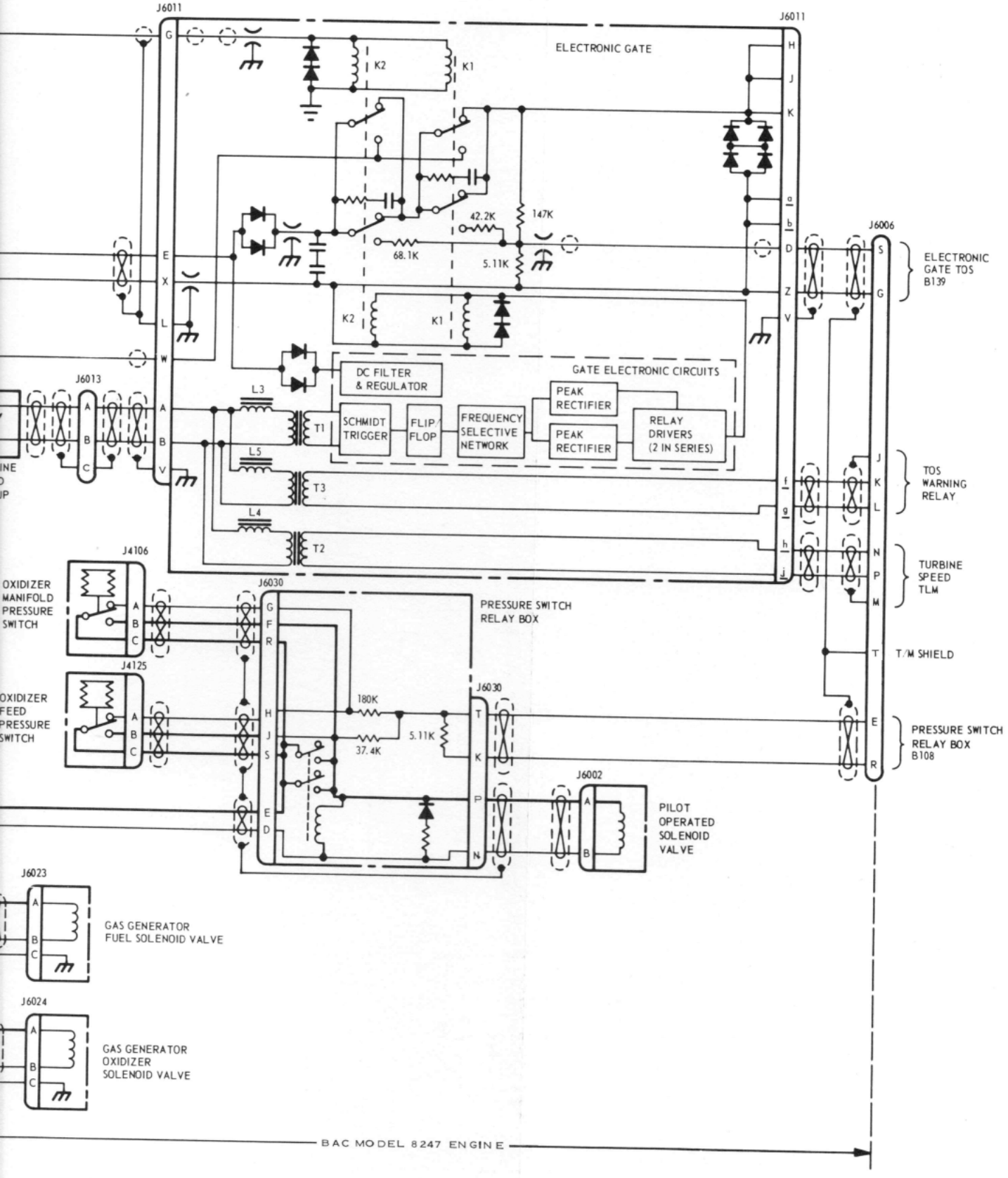
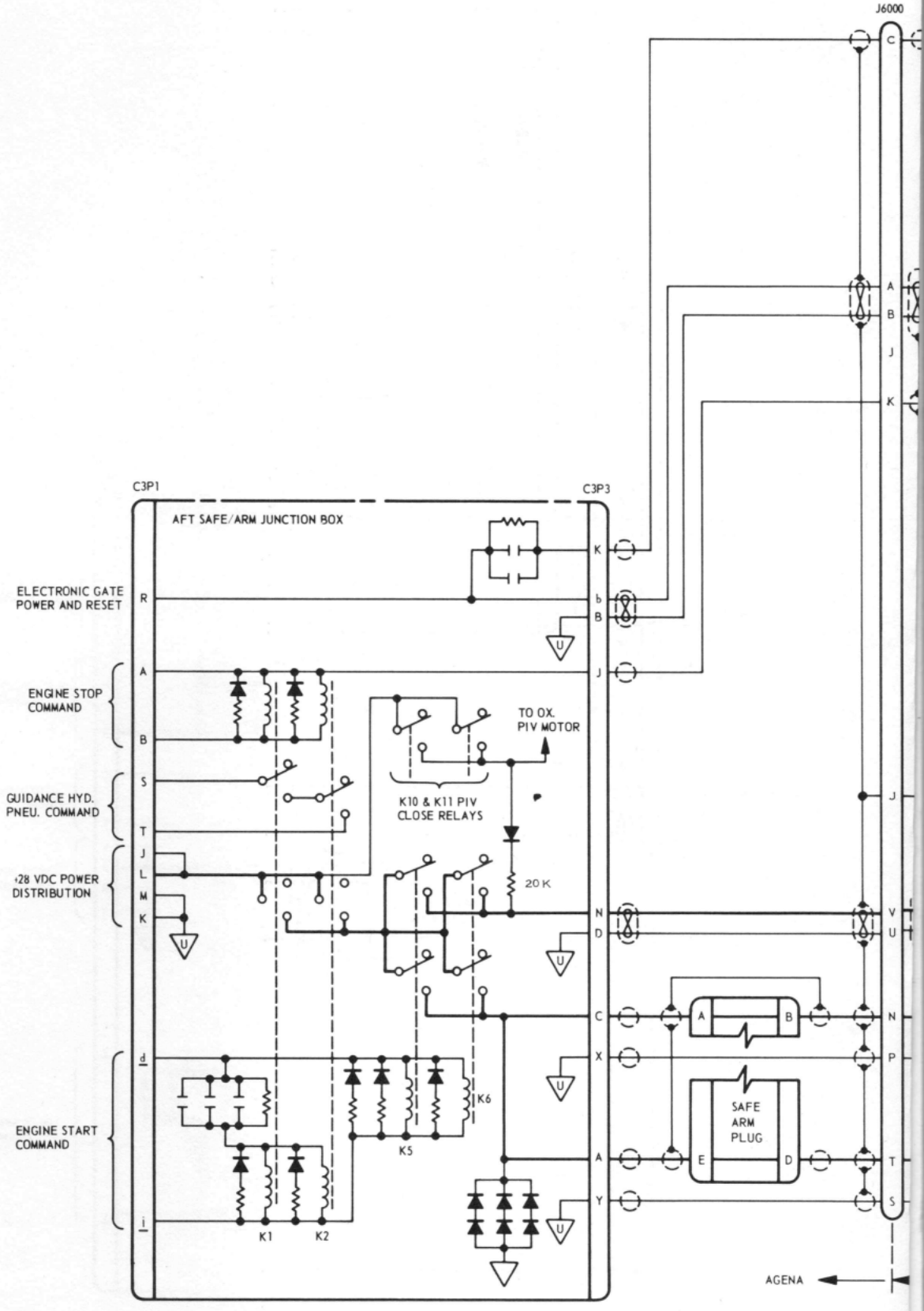


Fig. 2-3 Interim Engine Electrical Control Circuit


J6000
C3P1
C3P3
AFT SAFE/ARM JUNCTION BOX
ELECTRONIC GATE POWER AND RESET
ENGINE STOP COMMAND
GUIDANCE HYD. PNEU. COMMAND
+28 VDC POWER DISTRIBUTION
ENGINE START COMMAND
TO OX. PIV MOTOR
K10 & K11 PIV CLOSE RELAYS
20 K
K1
K2
K5
K6
SAFE ARM PLUG
AGENA

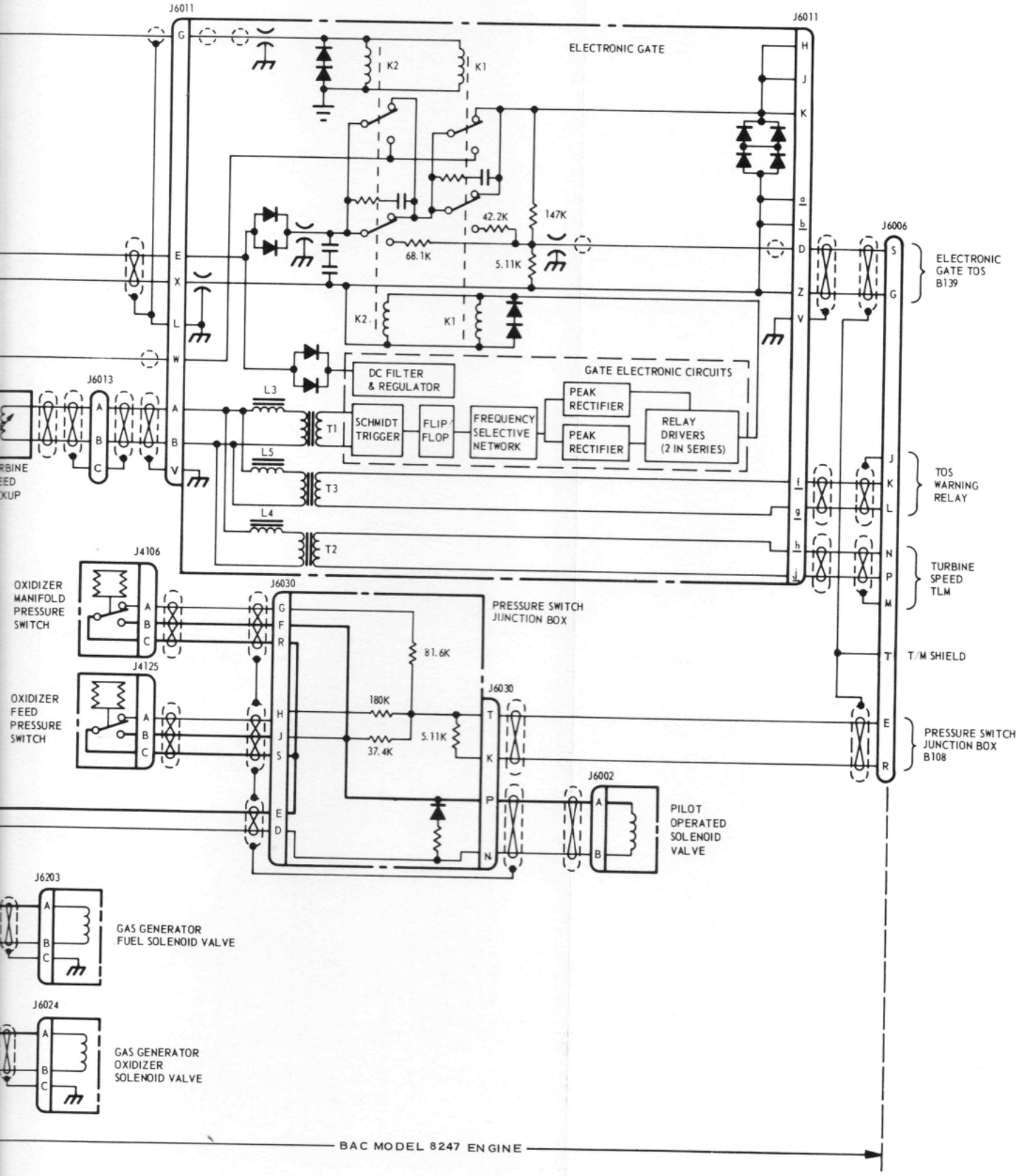
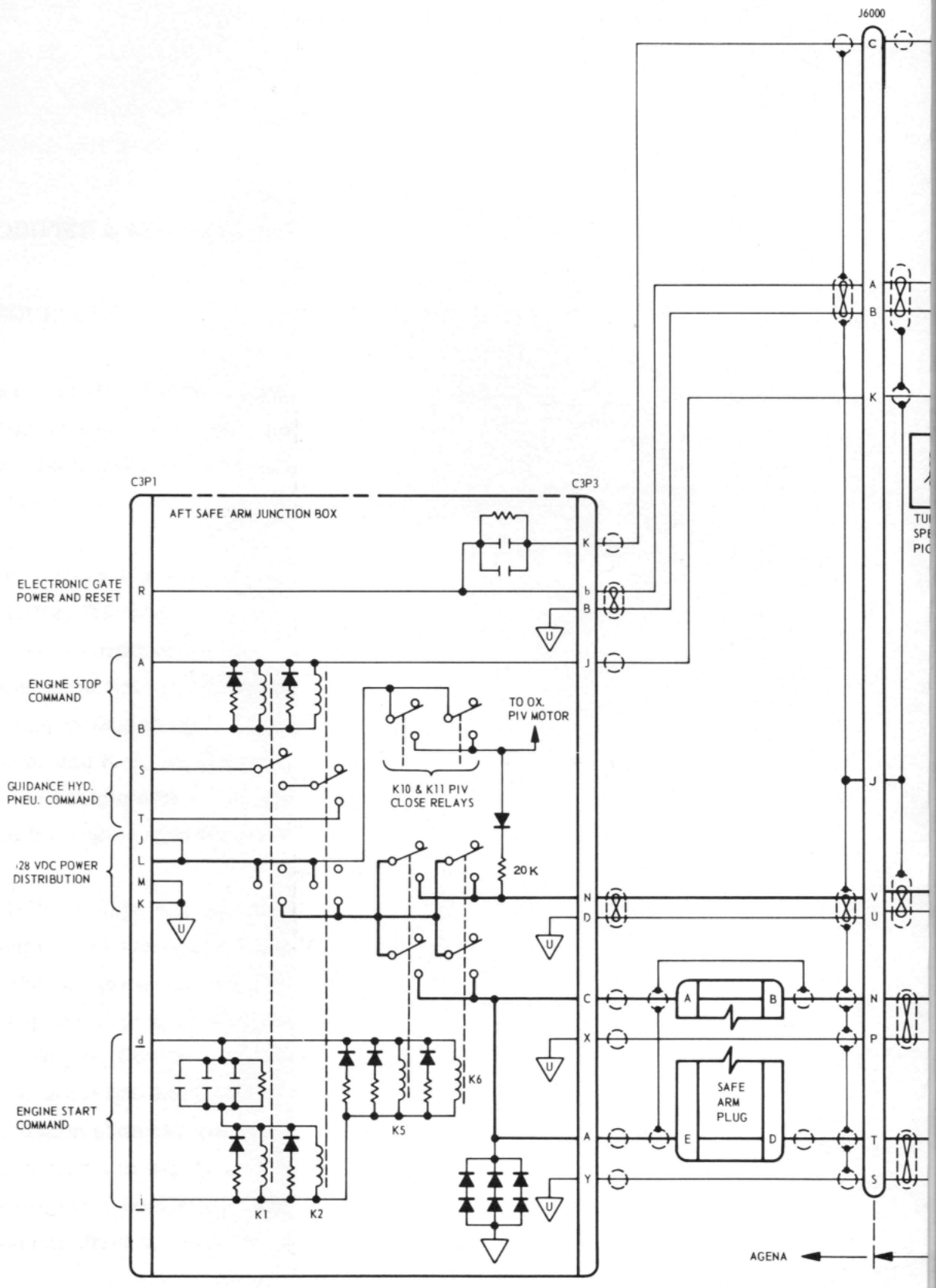


Fig. 2-4 Final Engine Electrical Control Circuit


J6000
C3P1
C3P3
AFT SAFE ARM JUNCTION BOX
ELECTRONIC GATE POWER AND RESET
ENGINE STOP COMMAND
GUIDANCE HYD. PNEU. COMMAND
28 VDC POWER DISTRIBUTION
ENGINE START COMMAND
TO OX. PIV MOTOR
K10 & K11 PIV CLOSE RELAYS
20 K
K1
K2
K5
K6
SAFE ARM PLUG
AGENA

Section 3
MODIFIED ENGINE START AND SHUTDOWN SEQUENCE

3.1 START SEQUENCE

The start sequence of the modified engine final configuration is shown graphically in Figure 3-1 and summarized in the following discussion. The summary makes reference to the electrical and propellant flow schematics shown in Figures 2-4 and 2-1, respectively

The engine start sequence is initiated by a vehicle electrical command signal to energize the K1, K2, K5, and K6 relays of the vehicle aft safe arm junction box (ASA J-Box). For on-orbit operations, electrical arm power is applied to the electronic gate at the same time power is applied to K-1, K-2, K-5, and K-6 and is maintained throughout engine operation until shutdown. Prior to the command signal to K-1, K-2, K-5, and K-6, an electrical pulse is applied to the electronic gate relay reset coils via the vehicle ASA J-box electrical pulse circuit to ensure that the electronic gate is in the engine operate mode.

When the ASA J-Box relays K1, K2, K5, and K6 are energized, electrical power is applied to the engine gas generator solenoid valves and the engine pressure switch junction box to the two pressure switches. The fuel and oxidizer gas generator solenoid valves open and propellants from the start tanks, under initial pressure of approximately 1000 psi, flow through the respective cavitating venturis to the gas generator. Hypergolic ignition initiates gas generator operation when the oxidizer enters the combustion chamber and impinges on the previously established fuel flow. The start tanks continue to supply propellants to the gas generator at gradually decreasing pressure until "boot strap" turbine speed is achieved. The gas generator combustion products drive a single-stage turbine rotor, accelerating the propellant

pumps through a gear box drive system. As the turbine speed increases, the propellant pump discharge pressures increase, and the main oxidizer valve opens at an oxidizer pump discharge pressure of approximately 130 psi and approximately 0.4 seconds after initiating the engine start sequence. During the ascent or first burn start, an engine oxidizer frangible disc ruptures at an oxidizer pump discharge pressure of approximately 180 psi. The turbine continues to accelerate as the oxidizer then flows through the oxidizer valve and thrust chamber regenative cooling passages. Also during the first-burn start, the fuel frangible disc ruptures at a fuel pump discharge pressure of approximately 550 psi.

Approximately 0.7 to 0.8 seconds after initiating the engine start sequence, boot strap turbine speed is obtained and the increasing propellant pump discharge pressures initiate propellant pump flow to the gas generator and recharge of the start tanks to the rated pressure for subsequent engine starts.

At approximately 0.85 seconds, the oxidizer pump discharge pressure at the oxidizer valve is sufficient, with the corresponding normal pressure perturbations, to cause momentary actuation of the oxidizer feed pressure switch (OFPS). At approximately 0.9 seconds, the thrust chamber regenerative cooling passages, oxidizer injector manifold, and oxidizer manifold pressure switch (OMPS) sensing line have been filled and the OMPS actuates, supplying electrical power to the pilot operated solenoid valve (POSV) via the pressure switch junction box. The oxidizer preflow into the thrust chamber combustion chamber at the time the OMPS actuates is about 2.0 pounds due to the time required to complete the OMPS pressure switch sensing line fill subsequent to establishing injector oxidizer manifold pressure.

The POSV is energized and closes, terminating fuel flow through the fuel valve actuation pressure and balance pressure return circuit to the fuel pump inlet. Terminating the fuel valve actuation pressure flow allows the actuation pressure to increase in the fuel valve actuation cavity and open the fuel valve. Approximately 10 milliseconds after the fuel valve starts to open, fuel flow into the thrust chamber combustion chamber is initiated through the four center fuel orifices of the injector

and impingement on the previously established oxidizer flow is initiated. Approximately 40 milliseconds later, the fuel injector manifold is filled and flow is established through the remaining fuel injector orifices. The thrust chamber combustion process is initiated by hypergolic reaction of the fuel (UDMH) and oxidizer (IRFNA). A nominal 6.0 to 7.0 pounds of oxidizer preflow occurs prior to filling the fuel injector manifold, after which thrust chamber ignition and thrust buildup occurs.

The thrust chamber combustion pressure results in a pressure increase at the OFPS sufficient to achieve positive actuation of the switch, which then terminates the chatter operation and completes the backup function to provide electrical power to the POSV. In the event of an OMPS fail-to-actuate malfunction, the turbine speed will continue to increase with a corresponding increase in pressure at the OFPS. Approximately 0.120 seconds after the nominal OMPS actuation time, the OFPS will actuate sufficiently to energize the POSV and initiate a thrust chamber operating sequence similar to that described above for the normal OMPS operation. An additional 4 to 6 pounds oxidizer preflow results from an OFPS-controlled start due to the additional time required to achieve the higher turbine speed.

3.2 SHUTDOWN SEQUENCE

Engine shutdown is accomplished by applying electrical power to the shutdown coils of ASA J-Box relays K1 and K2, thereby removing electrical power from the engine solenoid valves. The POSV opens, venting the fuel valve actuation pressure and allowing the fuel valve to close under spring and hydraulic load within 25 to 30 milliseconds, thus terminating fuel flow and consequently thrust chamber combustion. The direct-acting oxidizer and fuel gas generator solenoid valves close approximately 110 and 160 milliseconds, respectively, after removing electrical power. Gas generator solenoid valve closing terminates propellant flow to the gas generator, resulting in turbine speed decay. Approximately 1.5 to 2.0 seconds after the shutdown signal, turbine speed and corresponding oxidizer pump discharge pressure decay

sufficiently to allow the main oxidizer valve to close under spring load, terminating oxidizer post flow at approximately 29 pounds. In the event of a turbine overspeed greater than 29,500 rpm during engine operation with the electronic gate in the armed condition, the electronic gate relays are energized and send an electrical signal to the shutdown coils of the ASA J-Box relays K1 and K2, resulting in engine shutdown as described above.

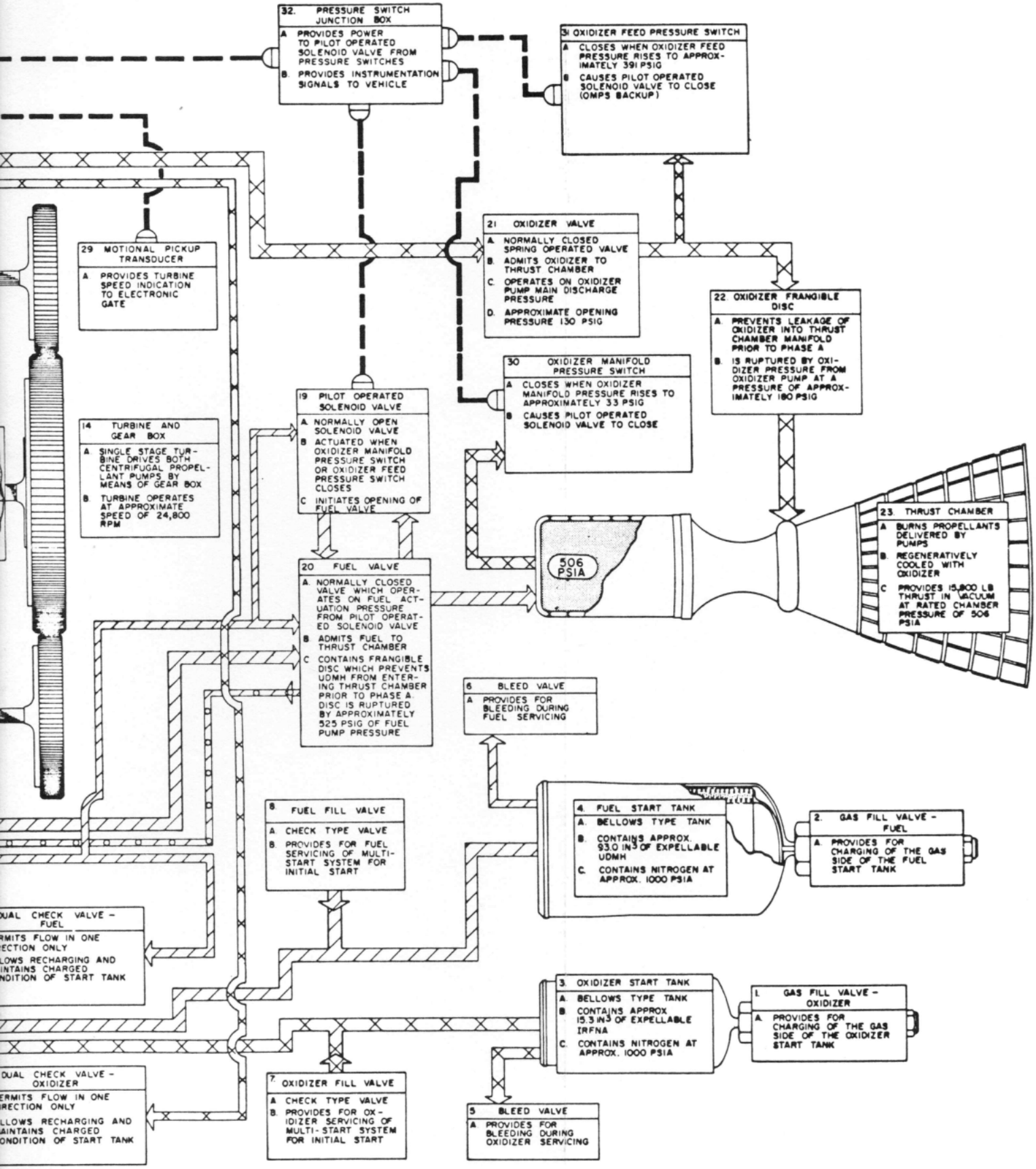
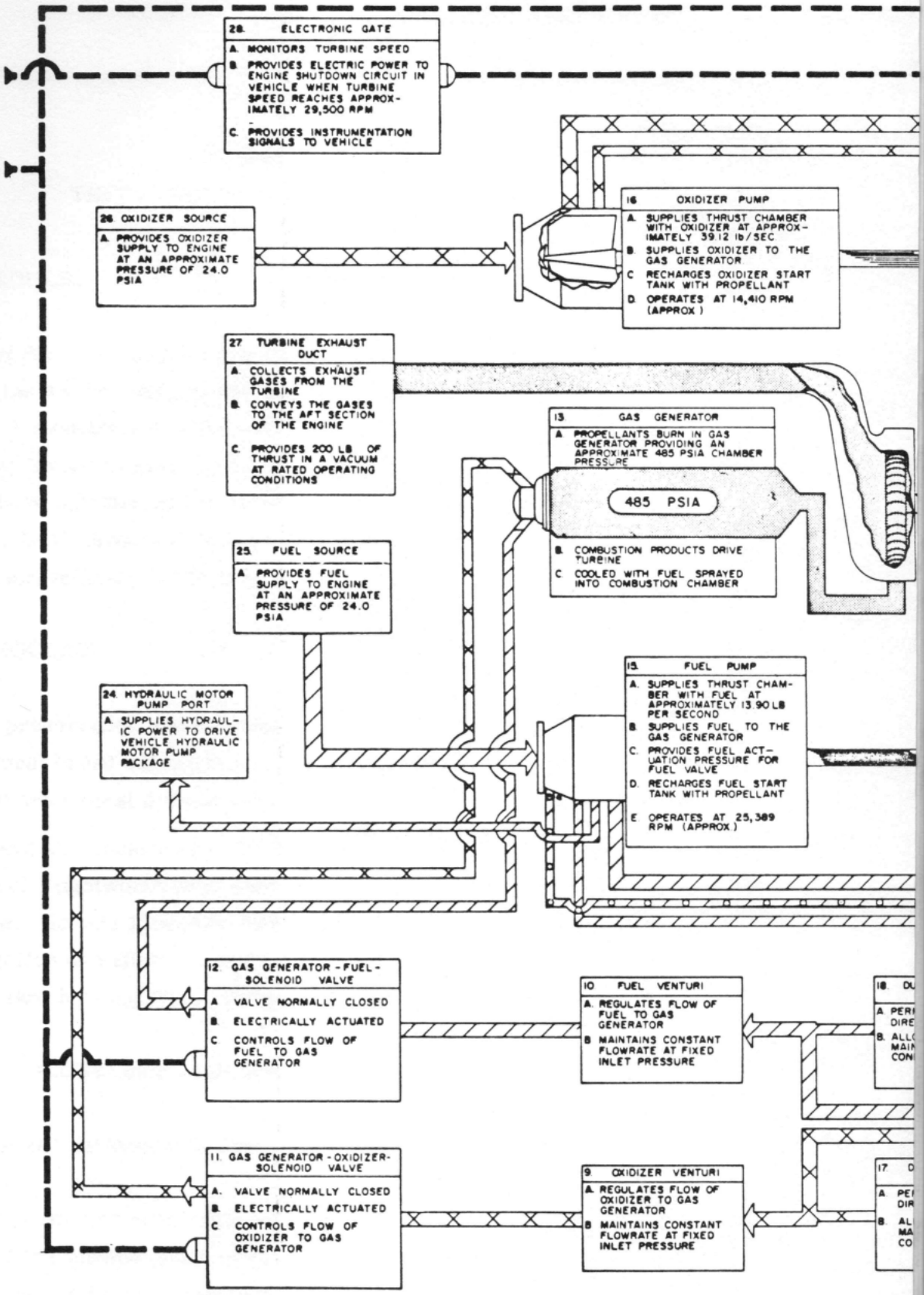

32. PRESSURE SWITCH JUNCTION BOX
A. PROVIDES POWER TO PILOT OPERATED SOLENOID VALVE FROM PRESSURE SWITCHES
B. PROVIDES INSTRUMENTATION SIGNALS TO VEHICLE
31 OXIDIZER FEED PRESSURE SWITCH
A. CLOSES WHEN OXIDIZER FEED PRESSURE RISES TO APPROXIMATELY 391 PSIG
B. CAUSES PILOT OPERATED SOLENOID VALVE TO CLOSE (OMPS BACKUP)
29 MOTIONAL PICKUP TRANSDUCER
A. PROVIDES TURBINE SPEED INDICATION TO ELECTRONIC GATE
21 OXIDIZER VALVE
A. NORMALLY CLOSED SPRING OPERATED VALVE
B. ADMITS OXIDIZER TO THRUST CHAMBER
C. OPERATES ON OXIDIZER PUMP MAIN DISCHARGE PRESSURE
D. APPROXIMATE OPENING PRESSURE 130 PSIG
22 OXIDIZER FRANGIBLE DISC
A. PREVENTS LEAKAGE OF OXIDIZER INTO THRUST CHAMBER MANIFOLD PRIOR TO PHASE A
B. IS RUPTURED BY OXIDIZER PRESSURE FROM OXIDIZER PUMP AT A PRESSURE OF APPROXIMATELY 180 PSIG
30 OXIDIZER MANIFOLD PRESSURE SWITCH
A. CLOSES WHEN OXIDIZER MANIFOLD PRESSURE RISES TO APPROXIMATELY 33 PSIG
B. CAUSES PILOT OPERATED SOLENOID VALVE TO CLOSE
19 PILOT OPERATED SOLENOID VALVE
A. NORMALLY OPEN SOLENOID VALVE
B. ACTUATED WHEN OXIDIZER MANIFOLD PRESSURE SWITCH OR OXIDIZER FEED PRESSURE SWITCH CLOSES
C. INITIATES OPENING OF FUEL VALVE
14 TURBINE AND GEAR BOX
A. SINGLE STAGE TURBINE DRIVES BOTH CENTRIFUGAL PROPELLANT PUMPS BY MEANS OF GEAR BOX
B. TURBINE OPERATES AT APPROXIMATE SPEED OF 24,800 RPM
506 PSIA
23. THRUST CHAMBER
A. BURNS PROPELLANTS DELIVERED BY PUMPS
B. REGENERATIVELY COOLED WITH OXIDIZER
C. PROVIDES 15,800 LB THRUST IN VACUUM AT RATED CHAMBER PRESSURE OF 506 PSIA
20 FUEL VALVE
A. NORMALLY CLOSED VALVE WHICH OPERATES ON FUEL ACTUATION PRESSURE FROM PILOT OPERATED SOLENOID VALVE
B. ADMITS FUEL TO THRUST CHAMBER
C. CONTAINS FRANGIBLE DISC WHICH PREVENTS UDMH FROM ENTERING THRUST CHAMBER PRIOR TO PHASE A. DISC IS RUPTURED BY APPROXIMATELY 525 PSIG OF FUEL PUMP PRESSURE
6 BLEED VALVE
A. PROVIDES FOR BLEEDING DURING FUEL SERVICING
8 FUEL FILL VALVE
A. CHECK TYPE VALVE
B. PROVIDES FOR FUEL SERVICING OF MULTI-START SYSTEM FOR INITIAL START
4. FUEL START TANK
A. BELLOWS TYPE TANK
B. CONTAINS APPROX. 93.0 IN³ OF EXPELLABLE UDMH
C. CONTAINS NITROGEN AT APPROX. 1000 PSIA
2. GAS FILL VALVE - FUEL
A. PROVIDES FOR CHARGING OF THE GAS SIDE OF THE FUEL START TANK
DUAL CHECK VALVE - FUEL
RMITS FLOW IN ONE ECTION ONLY
LOWS RECHARGING AND INTAINS CHARGED NDITION OF START TANK
3. OXIDIZER START TANK
A. BELLOWS TYPE TANK
B. CONTAINS APPROX. 15.3 IN³ OF EXPELLABLE IRFNA
C. CONTAINS NITROGEN AT APPROX. 1000 PSIA
1. GAS FILL VALVE - OXIDIZER
A. PROVIDES FOR CHARGING OF THE GAS SIDE OF THE OXIDIZER START TANK
DUAL CHECK VALVE - OXIDIZER
ERMITS FLOW IN ONE RECTION ONLY
LLOWS RECHARGING AND AINTAINS CHARGED ONDITION OF START TANK
7. OXIDIZER FILL VALVE
A. CHECK TYPE VALVE
B. PROVIDES FOR OXIDIZER SERVICING OF MULTI-START SYSTEM FOR INITIAL START
5 BLEED VALVE
A. PROVIDES FOR BLEEDING DURING OXIDIZER SERVICING

Fig. 3-1 Modified Engine Functional Block Diagram


28. ELECTRONIC GATE
A. MONITORS TURBINE SPEED
B. PROVIDES ELECTRIC POWER TO ENGINE SHUTDOWN CIRCUIT IN VEHICLE WHEN TURBINE SPEED REACHES APPROXIMATELY 29,500 RPM
C. PROVIDES INSTRUMENTATION SIGNALS TO VEHICLE
16 OXIDIZER PUMP
A. SUPPLIES THRUST CHAMBER WITH OXIDIZER AT APPROXIMATELY 39.12 lb/SEC
B. SUPPLIES OXIDIZER TO THE GAS GENERATOR
C. RECHARGES OXIDIZER START TANK WITH PROPELLANT
D. OPERATES AT 14,410 RPM (APPROX.)
26. OXIDIZER SOURCE
A. PROVIDES OXIDIZER SUPPLY TO ENGINE AT AN APPROXIMATE PRESSURE OF 24.0 PSIA
27 TURBINE EXHAUST DUCT
A. COLLECTS EXHAUST GASES FROM THE TURBINE
B. CONVEYS THE GASES TO THE AFT SECTION OF THE ENGINE
C. PROVIDES 200 LB. OF THRUST IN A VACUUM AT RATED OPERATING CONDITIONS
13 GAS GENERATOR
A. PROPELLANTS BURN IN GAS GENERATOR PROVIDING AN APPROXIMATE 485 PSIA CHAMBER PRESSURE
485 PSIA
B. COMBUSTION PRODUCTS DRIVE TURBINE
C. COOLED WITH FUEL SPRAYED INTO COMBUSTION CHAMBER
25. FUEL SOURCE
A. PROVIDES FUEL SUPPLY TO ENGINE AT AN APPROXIMATE PRESSURE OF 24.0 PSIA
15. FUEL PUMP
A. SUPPLIES THRUST CHAMBER WITH FUEL AT APPROXIMATELY 13.90 LB PER SECOND
B. SUPPLIES FUEL TO THE GAS GENERATOR
C. PROVIDES FUEL ACTUATION PRESSURE FOR FUEL VALVE
D. RECHARGES FUEL START TANK WITH PROPELLANT
E. OPERATES AT 25,389 RPM (APPROX.)
24. HYDRAULIC MOTOR PUMP PORT
A. SUPPLIES HYDRAULIC POWER TO DRIVE VEHICLE HYDRAULIC MOTOR PUMP PACKAGE
12. GAS GENERATOR - FUEL - SOLENOID VALVE
A. VALVE NORMALLY CLOSED
B. ELECTRICALLY ACTUATED
C. CONTROLS FLOW OF FUEL TO GAS GENERATOR
10 FUEL VENTURI
A. REGULATES FLOW OF FUEL TO GAS GENERATOR
B. MAINTAINS CONSTANT FLOWRATE AT FIXED INLET PRESSURE
11. GAS GENERATOR - OXIDIZER - SOLENOID VALVE
A. VALVE NORMALLY CLOSED
B. ELECTRICALLY ACTUATED
C. CONTROLS FLOW OF OXIDIZER TO GAS GENERATOR
9. OXIDIZER VENTURI
A. REGULATES FLOW OF OXIDIZER TO GAS GENERATOR
B. MAINTAINS CONSTANT FLOWRATE AT FIXED INLET PRESSURE

Section 4
TEST SCHEDULES AND CHRONOLOGY

4.1 TEST SCHEDULES

The testing effort directed toward evaluating and gaining confidence in the operation of the modified engine and its components commanded urgency and high priority. Full flightworthiness demonstration of the modified engine configuration was required, including both sea level and simulated altitude tests, prior to the flight of Vehicle 5003. To accomplish these objectives, top priorities were established with Air Force and NASA, at LMSC, BAC, Arnold Center and as required at other subcontractors, vendors, and commercial and military transportation facilities.

4.2 TEST CATEGORIES

For purposes of presentation and reference, the various phases of testing have been grouped into several broad categories. Test categories and the corresponding section within this report where test details are contained are as follows:

- Component and Subassembly Development Testing (Section 5)
- Sea Level Flightworthiness Testing (Section 6)
- Simulated Altitude Flightworthiness Testing (Section 7)
- Investigation of Failures and Discrepancies (Section 8)
- Special Development Tests (Section 9)

4.3 TEST CHRONOLOGY (See Table 4-1.)

4.3.1 Component and Subassembly Development Testing

These tests were conducted at the Bell Aerosystems Company in Niagara Falls, New York and included the turbine pump assembly, the gas generator start system, the electronic gate, and the pressure switch relay box.

Early in November 1965, test cell preparations were begun and the turbine pump assembly test series commenced on 12 November 1965. Initial planning indicated that a total of 61 starts would be required and that the program would be completed approximately 22 December 1965. As the program progressed it was found necessary to extend the testing to evaluate observed results and to ensure that sufficient data existed to affirm the durability and effectiveness of all components. This resulted in a total of 75 tests and a test series completion date of 9 March 1966.

Gas generator start system tests were initiated on 20 November 1965 and 30 tests were planned to comprise the series with completion scheduled for approximately 16 February 1966. As with the turbine pump testing, the results of the program dictated some additional testing with added, high response instrumentation. A resultant schedule extension occurred and a final total of 37 tests were concluded on 12 March 1966.

The electronic gate, POSV relay box, and engine electrical harness were subjected to elevated stress testing during the period January to March and the pressure switch relay box development tests were conducted during the period December 1965 to January 1966.

4.3.2 Sea Level Flightworthiness Demonstration Tests

This program was composed of three segments. One dealt with the component level pressure switch relay box qualification tests. The second portion covered the shock and vibration testing of the modified engine configuration, and a series of complete engine assembly test firings comprised the third segment. These activities were performed at the Bell Aerosystems Company utilizing engine XRM-4 (a non-fireable mass and center-of-gravity simulator) for the shock and vibration tests and engine 803 (which had been removed from Vehicle 5001 and modified) for the test firing series.

The pressure switch relay box FWD tests were initiated in January 1966 and successfully completed in February 1966. The test span for engine XRM-4 was planned for the period from 1 January to 15 January 1966; however, some hardware shortages and changes in instrumentation requirements resulted in delaying the start of testing until 22 January 1966. Completion of XRM-4 testing was achieved 7 February 1966.

Firings of engine 803 were planned to begin on 15 January 1966 and consisted of 43 starts in addition to the 5 required for acceptance after modification. Tests were to be completed by approximately 25 February 1966. Testing began essentially on schedule and continued until the test series was terminated on 14 February 1966 with three tests of the malfunction group remaining. This termination was dictated by the necessity to utilize engine 803 for simulated altitude testing at AEDC following the hard start of engine 805.

4.3.3 Simulated Altitude Flightworthiness Testing

Altitude testing of the XLR 81-BA-13 engine as modified under project Sure Fire was performed in test cell J2-A of the Rocket Test Facility at the Arnold Engineering Development Center, Arnold Air Force Station, Tennessee. This program, calling for a minimum simulated altitude of 250,000 feet, was planned to be accomplished in three phases: Phase I – Ignition Confidence tests, Phase II – Mission Simulation, and Phase III – Malfunction tests.

The schedule for this test activity was as follows: test cell occupancy on 8 December 1965, checkout tests on 21 February, Phase I from 1 March to 5 March 1966, Phase II from 12 March to 16 March 1966, Phase III from 29 March to 2 April 1966, and test cell occupancy terminated by 8 April 1966.

The cell was occupied on schedule and the preparation of the facility systems together with the installation and instrumentation of the engine were accomplished on an accelerated basis. This resulted in being ready for checkout tests on 7 February 1966, a schedule

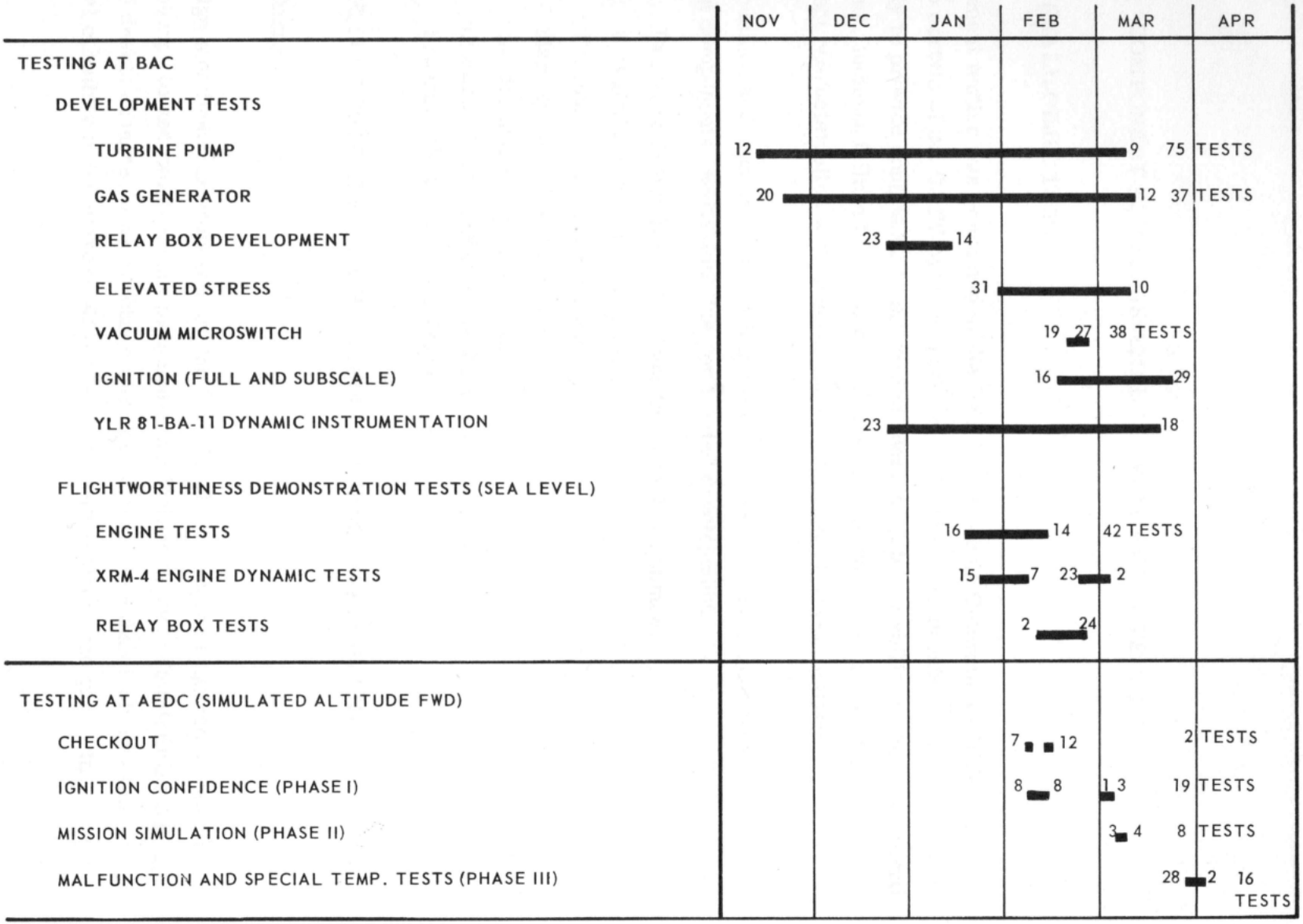
improvement of two weeks. Further schedule compression was planned by combining the checkout tests with Phase I, and testing was begun accordingly. Six tests were completed successfully; however, on the seventh test, a "hard start," subsequently attributed to water and alcohol contamination of the fuel system, resulted in failure of engine 805 on 12 February 1966.

As a result of this failure, engine 803 was released from the sea level flightworthiness demonstration test, refurbished, and delivered to AEDC. Engine 803 was installed on 18 February and testing was resumed on 1 March 1966.

In order to recover the schedule time lost as a result of engine 805 failure, the balance of Phase I was modified and combined with Phase II. These combined tests, 22 in number, were completed on 4 March 1966. Following this series of tests the engine and systems were cleaned, functional tests of the engine were conducted, and refurbishment and/or replacement of certain components was accomplished. The Phase III tests were redefined in the light of data from the earlier tests and testing for this final phase was begun on 28 March 1966. After two tests were completed, instrumentation difficulties were noted which required two days of down time for corrective action. Following this, the series was resumed on 30 March. On 1 April a predicted hard start occurred during a fuel lead test causing engine damage, including rupture of the oxidizer line between the main oxidizer valve and the thrust chamber oxidizer inlet manifold. This was the sixteenth test of the Phase III series of 22 and marked the conclusion of the program. Removal of the engine and cleanup of the facility was achieved by 8 April, meeting the original schedule established in November 1965.

Table 4-1

TEST CHRONOLOGY



Section 5
COMPONENT AND SUBASSEMBLY DEVELOPMENT TESTING

5.1 OVERALL OBJECTIVES

Development testing was conducted at the Bell Aerosystems Company facilities to evaluate certain of the GATV primary propulsion system components. The tests were generally to provide confidence in the design, demonstrate reliability under operational conditions, indicate failure levels in overstress, and to compare modified engine component characteristics with those of the proven YLR 81-BA-11 engine.

The discussion and data provided in this section covers testing conducted for the following components, subassemblies, or related development:

- Pressure Switch Relay Box/Junction Box Development
- Elevated Stress Test Program
- Turbine Pump Test Program
- Start System Test Program
- Special Instrumentation Tests
- Oxidizer Feed Pressure Switch Durability Tests
- Pressure Switch Vacuum Tests

5.2 PRESSURE SWITCH RELAY BOX/JUNCTION BOX DEVELOPMENT

5.2.1 Design History

The design modifications proposed for the XLR 81-BA-13 engine required a control circuit change to use two pressure switches and a lock-in relay. To minimize the required design changes, a new package was developed rather than including the additional circuitry in the existing XLR 81-BA-13 engine electronic gate. In

November 1965, BAC started design of a new engine control package designated the pressure switch relay box (PSRB). The PSRB incorporated a lock-in relay, telemetry status network, pilot operated solenoid valve EMI suppression, and interconnect wiring for connecting the pressure switches and relay into the engine harness.

On 3 December 1965 changes were made in the engine control circuit involving the use of the electronic gate which required a new engine harness. Analysis of early engine operation had indicated a bias on the telemetry (TM) signals caused by the high engine-current levels in the common power and TM ground return wiring. To eliminate this bias in the PSRB TM data, a separate TM system ground return was included in the new harness and the PSRB was changed to provide a floating TM output. A schematic diagram of the PSRB is shown in Fig. 5-1a.

On 27 December 1965 the PSRB failed during development testing when the relay remained transferred after electrical power was removed. Development testing was continued on the PSRB (-1), however, during investigation of the relay failure. The -1 PSRB was in final development test when on 21 January 1966, Air Force direction to eliminate the relay was received. Relay elimination required a redesign of the PSRB at a point in the program schedule when two flight engines had been shipped to LMSC, the BAC sea level testing was underway, and the AEDC altitude testing was two weeks away from first firing. To avoid complete redesign of the PSRB, BAC was directed to utilize the PSRB design, eliminating only the relay (Fig. 5-1b). Internal and external structure was unchanged to enable use of already fabricated subassemblies and to simplify vehicle and test engine retrofit.

Figure 5-2 shows the internal configuration differences of the PSRB and the new pressure switch junction box (PSJB). It was necessary to have the first of the new PSJB units ready by 23 January for shipment to AEDC for use in the altitude testing program. One PSJB unit, identified as BAC P/N 8247-472383-1, was assembled and tested on 23 January and shipped to AEDC on schedule. At the same time, the effects of the PSRB redesign were being reviewed at LMSC. From the design evaluation undertaken and the turbine pump and sea level FWD engine tests, it was concluded

that a knowledge of the operation of the oxidizer feed pressure switch (OFPS) and oxidizer manifold pressure switch (OMPS) was important in analyzing and verifying a proper engine start transient sequence. The TM network in the original PSRB did not distinguish between the operation of the OMPS and OFPS as shown in Figure 5-3. BAC was directed to include another TM network resistor which would distinguish between the individual actions of the two switches.

The TM network addition could not be incorporated in the first PSJB (-1) under assembly at that time for the AEDC test program. Resistance values were chosen and fabrication of a -7 design, Figure 5-1c, began on 26 January for PSJB units to retrofit the flight vehicle at the launch facility. The -1 PSJB sent to AEDC was used for the first seven altitude tests. When engine 803 was delivered to AEDC for the balance of the test program, the -7 PSJB was used.

It was recognized during analysis of the PSRB redesign, that the results of the test programs then underway might dictate a return to the lock-in relay design. Therefore, a new PSRB (-9) was designed using a high-reliability -3 relay. The -9 PSRB was identical in all respects with the -1 PSRB except for the controlled-quality relay. Two of these PSRB units, BAC P/N 8247-472383-9, were assembled and readied for use if the design requiring the lock-in relay concept became necessary.

No further design changes were made in either the PSRB or PSJB. The final flight configuration engine control system utilized the -7 pressure switch junction box.

In summary, four distinct pressure switch relay box/junction box configurations were designed and built:

- Relay box (-1), original relay design (Figure 5-1a)
- Junction box (-1), original J-box design with 3-level TM readout (Figure 5-1b)
- Junction box (-7), 4-level TM readout (Figure 5-1c)
- Relay box (-9), 3-level TM readout with high-reliability relay (Figure 5-1a)

All four designs used the same structural assembly, same production techniques, and same electrical components and wiring, except for the relay and TM resistor in the -9 and -7 configurations respectively. Development tests and subsequent component FWD and elevated stress tests were conducted using the -1 configuration pressure switch relay box.

5.2.2 Development Testing

The pressure switch relay box development tests provided design engineering evaluation prior to assembly of production units for qualification and flight use. The tests were conducted at the BAC facilities from 23 December 1965 through 14 January 1966. Test particulars are shown in Table 5-1.

Test Configuration. Two 8247-472483-1 relay boxes were used. The test configuration was identical to that used for the FWD component level tests (Section 6). The relay boxes tested were the first units produced and were considered to be representative of manufacturing and inspection practices which would be employed for flight production units.

Test Results. During vibration testing of the first relay box, the relay stuck in the energized position when the power was removed. Testing of the failed unit was suspended and an analysis was conducted to determine the cause of the failure (Section 8). The analysis disclosed that the conditions under which the relays were assembled were not sufficiently controlled to prevent contamination of the relay mechanism; the most probable cause of failure. The problem did not recur during subsequent testing of a second -1 relay box.

During vibration testing of the second -1 relay box, the bonding straps utilized on the shock mounts failed. The straps were redesigned using a larger bend radius to reduce stress. No problems were encountered during subsequent qualification testing (Section 6) with the new strap design.

Conclusions. Two problems (stuck relay and failed bonding straps) were encountered during development testing of the PSRB. Design changes provided correction of these problems as demonstrated by subsequent component FWD and elevated stress testing.

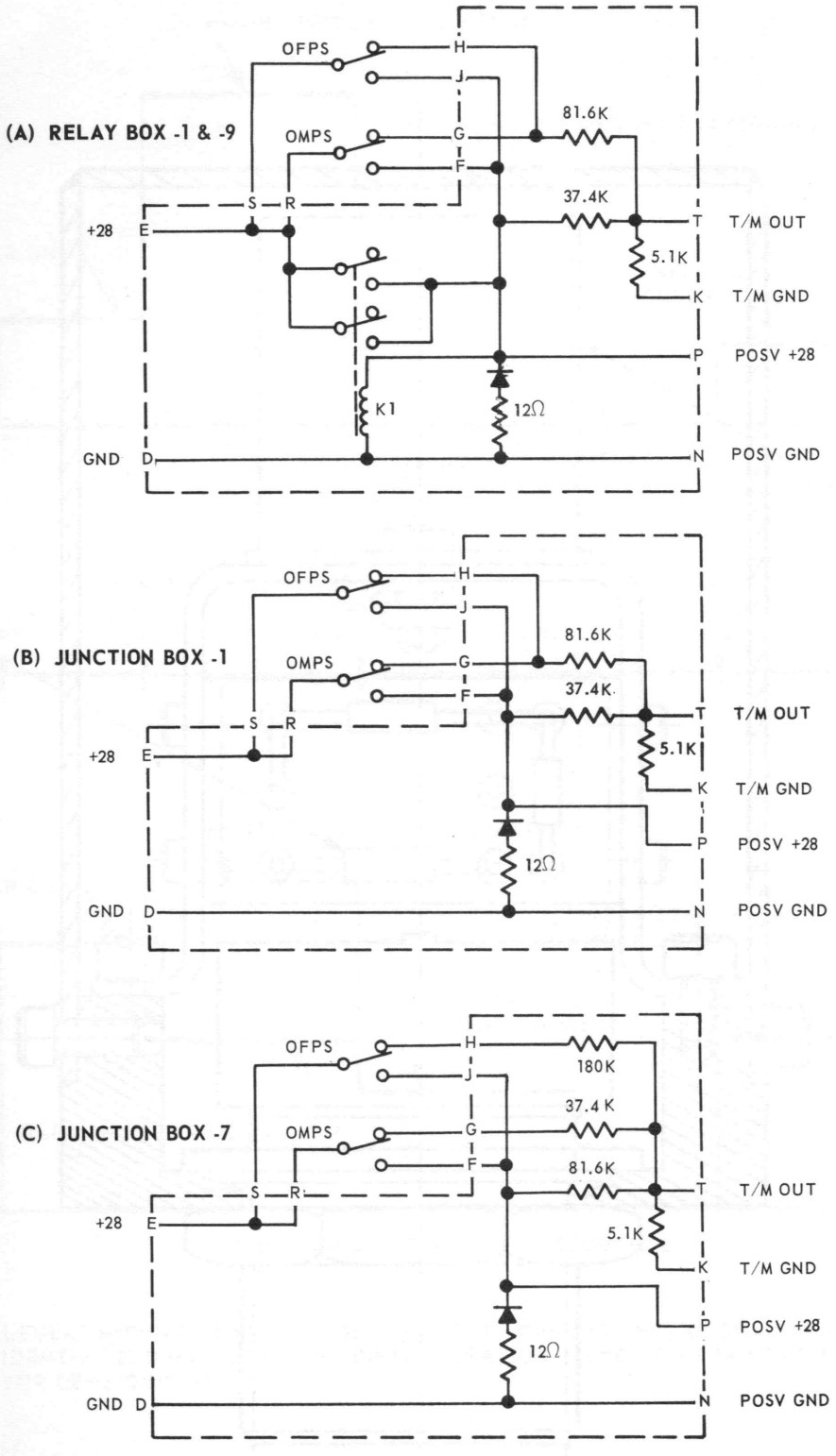


Fig. 5-1 Relay Box/Junction Box Circuit Diagrams

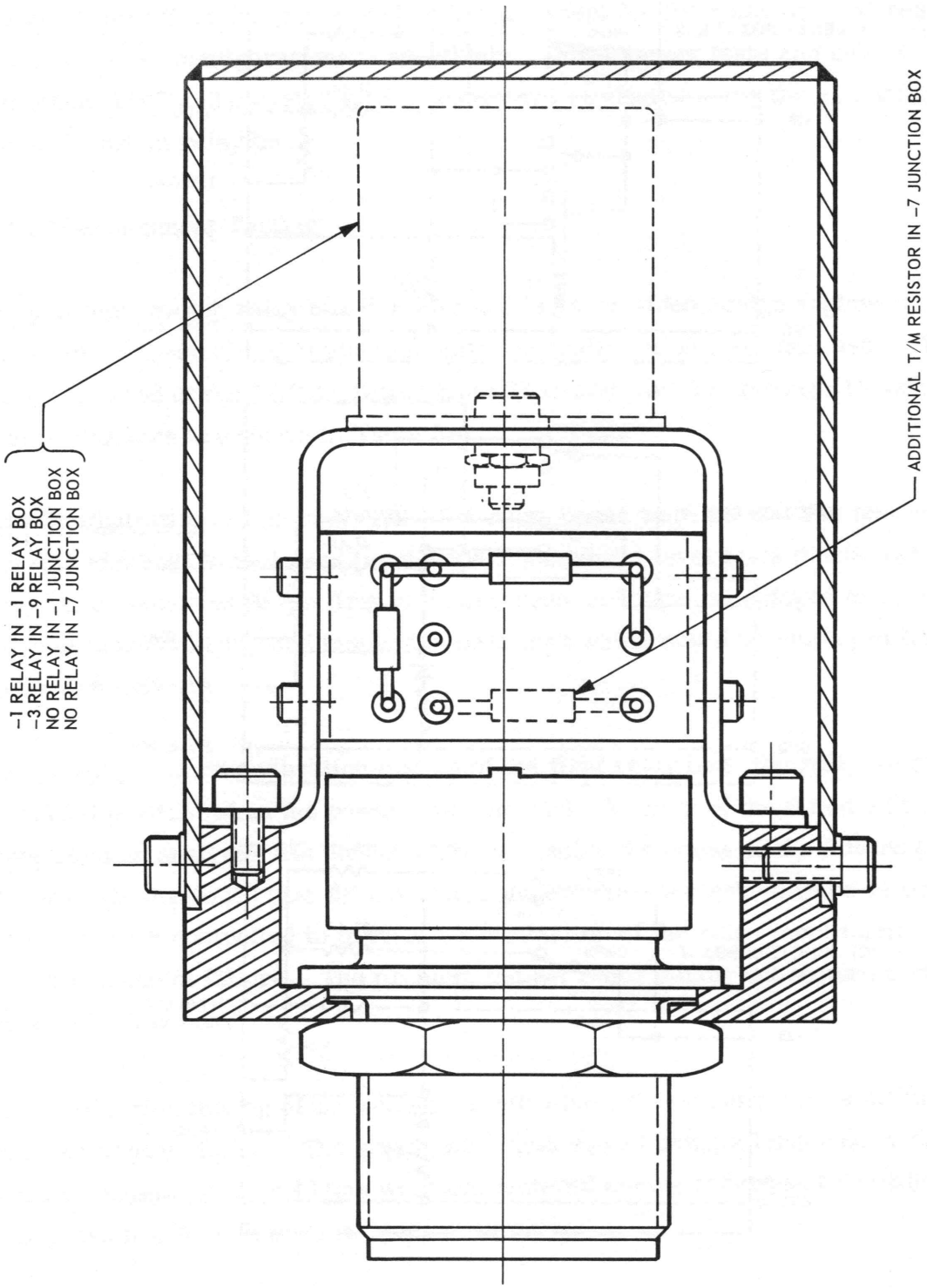
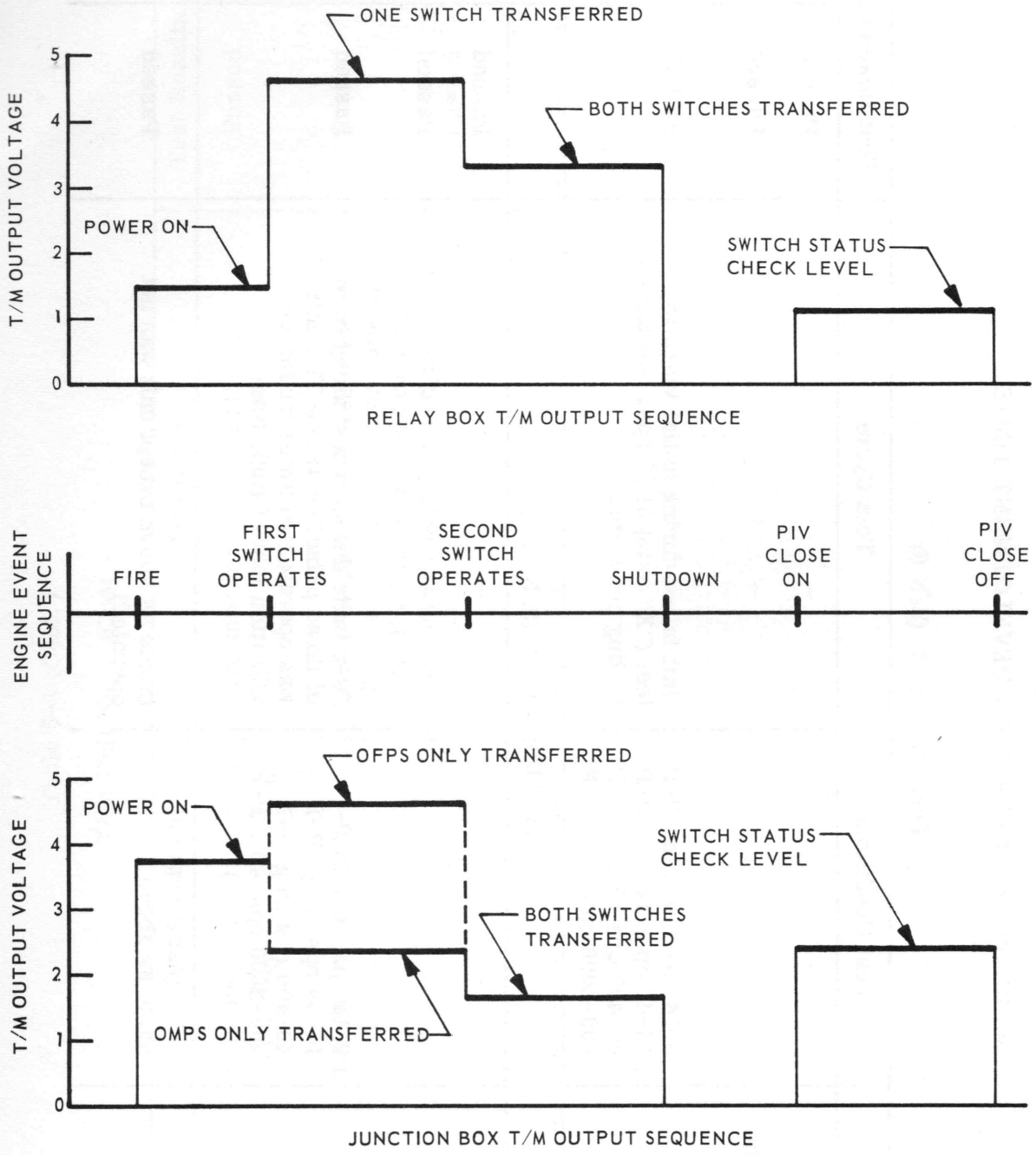

-1 RELAY IN -1 RELAY BOX
-3 RELAY IN -9 RELAY BOX
NO RELAY IN -1 JUNCTION BOX
NO RELAY IN -7 JUNCTION BOX
ADDITIONAL T/M RESISTOR IN -7 JUNCTION BOX

Fig. 5-2 Relay Box/Junction Box Comparison



NOTE: LEVELS SHOWN FOR NOMINAL 28 VDC INPUT. ORIGINAL RELAY BOX LEVELS DO NOT IDENTIFY SEQUENCE OF OFPS/OMPS OPERATION. JUNCTION BOX HAS DISTINCT LEVELS FOR OFPS/OMPS SEQUENCE.

Fig. 5-3 Relay Box/Junction Box Telemetry Levels

Table 5-1

PRESSURE SWITCH RELAY BOX DEVELOPMENT TESTS

Test Unit No. 1 (S/N 6)

Test	Test Conditions	Test Cycle	Test Results
Functional Test	–	–	Passed
Insulation Resistance and Dielectric Strength	–	–	Passed
Sine Vibration – Operating	5-14 cps 0.5 in. P-P 14-40 cps 10 g P-P 40-400 cps 15 g P-P 400-3000 cps 40 g P-P	Unit failed during initial vibration test ("X" axis) at 37 cps. No further testing conducted	Failed
Test Unit No. 2 (S/N 7)			
Functional Test	–	–	Passed
Insulation Resistance and Dielectric Strength	–	–	Passed
Sine Vibration – Operating	5-14 cps 0.5 in P-P 14-40 cps 10 g P-P 40-400 cps 15 g P-P 400-3000 cps 40 g P-P	30-minute duration test along each of three principle axes. The unit was operated 10 times during the vibration test of each axis.	Passed
Sine Vibration – Non operating	Same as above.	(Same as above except unit was not operated)	Passed

Table 5-1 (Cont.)

Test Unit No. 2 (Cont.)

Test	Test Conditions	Test Cycle	Test Results
Random Vibration	20 to 400 cps 0.08 g^2/cps 400 to 2000 cps 0.2 g^2/cps	Five minutes on each of these principle axes. Unit operated 10 times each run.	Passed
Humidity Test	95% RH	Three temperature cycles between +35° F and +110° F. Three days at +110° F followed by insulation resistance and dielectric strength tests.	Passed
Insulation Resistance and Dielectric Strength		–	Passed
Functional Test	–	–	Passed

5.3 ELEVATED STRESS TEST PROGRAM

The basic objective of the elevated stress test program was to determine the operational limits of specific engine electrical control system components under various conditions of environmental and operational overstress. Secondary objectives were to evaluate component degradation as stress levels increased and to identify the failure cause should a component fail at a stress level less than the established test limits. The test program included three primary engine electrical components: electronic gate, relay box, and engine harness. All testing was conducted at BAC between 31 January 1965 and 10 March 1966.

5.3.1 Failure Prediction Analysis

To determine the optimum test program, a failure prediction analysis was first undertaken prior to actual testing. The purpose of the analysis included the following requirements:

- Selection of optimum test environments and levels
- Selection of the most susceptible component operating mode for each environment
- Selection of the optimum test sequence

LMSC 1414463, the Model Specification for the XLR 81-BA-13 Engine, and LMSC 6117D, General Environmental Specification for Equipment of the Agena and Associated Payload, were used to determine the environmental conditions that would reflect actual service usage. The relative sensitivity of the components to the various environmental conditions, and to the vehicle interface conditions, was assessed to determine which of the environments and interface conditions presented the most severe test of component design margins. A summary of the analysis and the test conditions selected for the elevated stress test program are shown in Table 5-2.

The failure prediction analysis provided a technical base for determining the combination of environmental stress and operating conditions which would best demonstrate the operational limit and margin of the component on a worst-on-worst basis. The analysis resulted in reduced testing requirements and increased significance of results, both of which were considered critical under the accelerated Sure Fire schedule.

5.3.2 Test Plan

The test program was conducted utilizing a special functional test procedure designed for each component. The special functional test procedure for the electronic gate is shown in Table 5-3. These tests were performed each time the functional test was called out in the stress testing sequence and all data were recorded. Special functional tests comparable to those in Table 5-3 were also designed for the relay box and engine harness. Test environments and conditions for the electronic gate and relay box are shown in Table 5-4; for the engine harness in Table 5-5. A typical sequence of testing, using the special functional test, is shown in Figure 5-4. This sequence is typical for the overall shock and vibration test program. For the temperature tests, a unit to be tested was stabilized at each temperature and the special functional test then performed. For the voltage tests, the supply voltage was first set to the stress level requirement and then the special functional test was performed.

The special functional test was scheduled for performance over 75 times under as many as 12 levels of 4 different environments for each component. Through the use of the special functional test, the performance of the unit would be accurately compared as it was sequentially subjected to increased levels of test. Degradation of the unit performance as stress increased would thus be measured and quantitatively assessed.

All test components used were assembled under normal manufacturing conditions, subjected to standard inspection, and acceptance tested. Flight-type shock mounts were used during the testing and all test harnesses were assembled from flight-type material.

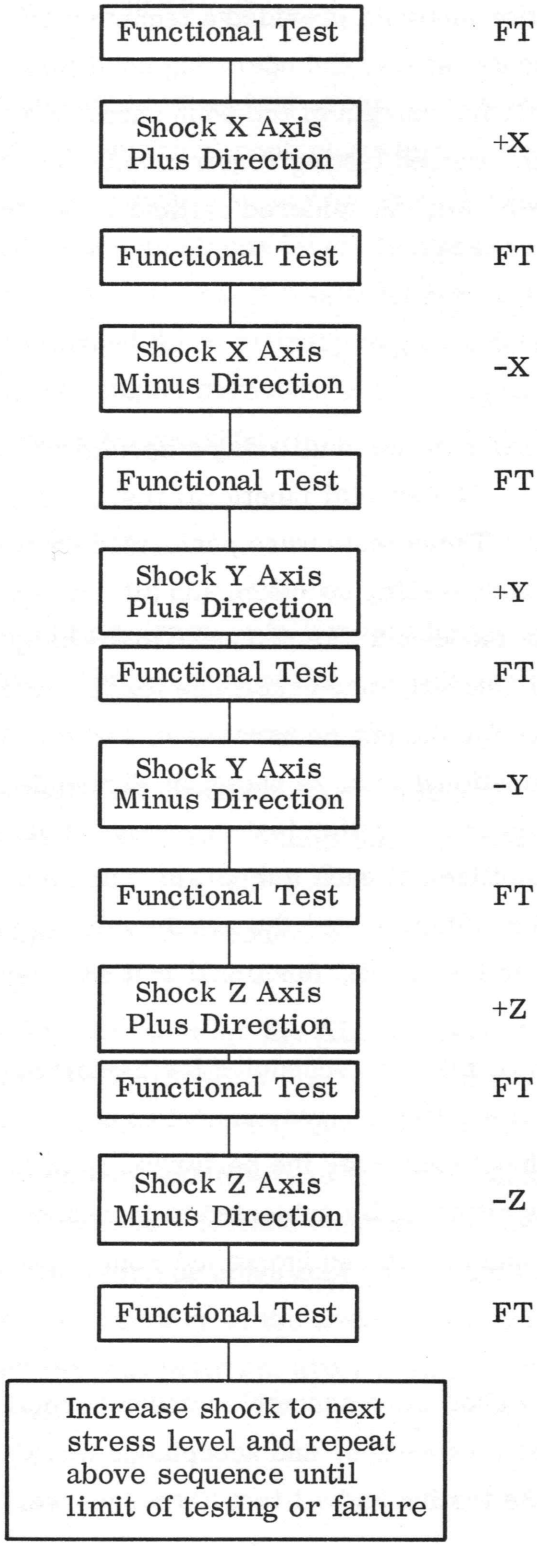


Fig. 5-4 Typical Overstress Test Sequence

Distinction between failure and out-of-specification operation was made in the elevated stress test plan and procedures. The basic intent of the elevated stress testing was to determine the operational limit of the component under conditions of stress in excess of the specification limits. Therefore, the primary definition of operational limit was that point at which the unit was incapable of performing its intended function without regard to specification values for operating parameters. For the electronic gate, the ability to at least trip and reset regardless of the level of input/output conditions was considered as meeting its intended function. For the relay box, the ability to at least latch and unlatch the lock-in relay regardless of level of input/output conditions was considered as meeting its intended function. For the engine harness, the ability to maintain continuity of operating circuits, i.e., the non-telemetry wiring, was considered as meeting its intended function. A secondary objective was to determine degradation of the components under test. This was defined as the area of operation in which the unit met the above defined intended function but would not demonstrate complete compliance with specification requirements. In addition, if the unit did not meet its intended function under elevated stress conditions but did so under nominal conditions, it was considered operable and testing at the same elevated stress or at a reduced level of stress was conducted to determine the repeatability of the failure. If, however, the unit failed to meet its intended function under nominal conditions after having failed under stress, the unit was considered permanently inoperable and testing was terminated.

5.3.3 Test Results

The summary of test results (Table 5-6) is presented in terms of the major factors defined in paragraph 5.3.2: (1) overstress range wherein the component met its intended function, (2) overstress range wherein the component demonstrated operation within specification, and (3) overstress limit at which the component became permanently inoperable. The operating conditions and associated ranges and limits for each component tested are discussed below.

Electronic Gate. The functional limit of the electronic gate was reached prior to achieving the planned upper test temperature limit of +400° F. At temperatures above +280° F the gate would not reset reliably. Below -40° F and above +200° F, the lower trip frequency and trip bandwidth were out of specification limits. The low-voltage test requirement was to determine the minimum voltage at which the gate would operate. This limit was reached at 13 VDC at which voltage the gate would not trip. Below 17 VDC the trip bandwidth was out of specification. The gate operated within specification to the upper test limit of 32 VDC which was imposed because the 32-volt zener diodes would fail in a predictable manner at higher voltages. No limits on duty cycle or shock were found within the test ranges of 20 days duty cycle and 160 g's shock.

Relay Box. The functional limit of the relay box was reached upon achieving the upper test temperature of +400° F. At this temperature, an internal solder joint at the connector melted and resulted in an open circuit at the +28V input. Disassembly of the relay box showed a general condition of melting and recrystallization of the solder joints throughout the unit. At temperatures above +320° F, the relay would not operate reliably at the specification minimum voltage of 19.5 VDC. Operation downward to -80° F remained within specification. Below a voltage of 12.8 VDC the relay would not operate; however, the box operated from 12.8V to the test limit of 42.5 VDC and performed within specification throughout. No limits on vibration or shock were found within the test limits of 37.5 g (RMS) random vibration and 160 g's shock.

Engine Harness. Inspection of the engine harness following the +400° F temperature test disclosed general solder melting and recrystallization, however, no open circuits or shorts occurred during the testing. Leakage current was less than 500 microamps in all tests performed and insulation resistance was greater than 100,000 megohms in all tests except for the high temperature tests wherein the lowest reading was in excess of 100 megohms.

5.3.4 Conclusions

The electronic gate demonstrated operation in accordance with specification requirements within the range of ±40° F beyond specification requirements and functional capability within the range of -80 to +120° F beyond specification requirements. The gate demonstrated operation in accordance with specification requirements within the range of ±2 volts beyond specification requirements and -5, +2 volts beyond specification requirements before functional capability was lost. The gate withstood shock levels four times the qualification level and operated at the temperature extremes with electrical power applied for four times the nominal mission life. Based on the above results, the electronic gate exhibits adequate design and operational margins for the planned Gemini missions.

The relay box and electrical harness demonstrated operation in accordance with specification requirements at temperature and voltage extremes greater than the electronic gate. Consequently the relay box and harness have design and operational margins at least consistent with the electronic gate, and are more than adequately designed for the planned Gemini missions.

Table 5-2

ANALYSIS OF TEST CONDITIONS

Item	Environment	Analysis	Test	
			No	Yes
1	Acceleration	Stress negligible compared to shock and vibration.	X	
2	Acoustic Excitation	Adequately demonstrated in numerous ground and flight engine firings.	X	
3	Corrosive Atmosphere	Same as item 2	X	
4	Explosive Atmosphere	Adequately demonstrated in production and qualification testing.	X	
5	Fungus	Not applicable, as cited in paragraph 1.4b and 4.10.3 of LMSC 6117D.	X	
6	Humidity	Adequately demonstrated in production and qualification testing.	X	
7	Pyrotechnic Shock	Excluded from elevated stress testing. Tests conducted by LMSC at Santa Cruz Test Base.	X	
8	Radiation	Adequately demonstrated in flight (C. F. paragraph 4.10.13 of LMSC 6117D).	X	
9	Rain, Salt, and Fog	Rain applies only to unsheltered equipment. Units are encapsulated. (C. F. paragraph 4.10.6 of LMSC 6117D and USCN 2R1 page 18 of LMSC 6117D).	X	
10	Sand and Dust	Not applicable, as the only moving parts are relays which are hermetically sealed.	X	
11	Shock	Units possibly subject to adverse effects at elevated stress levels.		X
12	Temperature	Same as item 11		X

Table 5-2 (Cont.)

Item	Environment	Analysis	Test	
			No	Yes
13	Vacuum	Arcing or corona is not a problem in hermetically sealed relays. Voltage levels are low in harness. Encapsulation and potting materials adequately demonstrated in vacuum.	X	
14	Vibration	Same as item 11		X
15	Voltage Levels	Same as item 11		X
16	Load Currents	Load currents are 10 to 50 times less than the individual piece part ratings of components directly connected to or through the interface (relays, connectors, etc.).	X	
17	Mounting and Structure	Subject to adverse effects under dynamic loads. Shock and vibration testing (items 11 and 14 above) will be conducted with flight mounting and cable installation.		X
18	Duty Cycle	Duty cycle of vehicle configuration could exceed original design considerations for Electronic Gate.		X

Table 5-3

ELECTRONIC GATE SPECIAL FUNCTIONAL TEST

Sequence	Standard Conditions	Variable Condition	Test Objective
Test 1 Trip Test	DC Supply = 19.5 & 30.5 VDC Turbine Output = 1.4 V P-P	Turbine Frequency	Verify that gate trips at standard conditions
Test 2 Reset Test	DC Supply = 19.5 & 30.5 VDC Turbine Output = 1.4 V P-P Turbine Frequency = 970 cps	(None)	Verify that gate resets at standard conditions
Test 3 Reset Pulse Level	Turbine Output = 1.4 V P-P Turbine Frequency = 970 cps	DC Supply Voltage	Determine minimum DC supply level at which gate can be reset with vehicle type pulse network
Test 4 Trip Frequency (Average of 3 runs)	DC Supply = 19.5 VDC Turbine Output = 1.4 V P-P	Turbine Frequency	Determine upper and lower frequencies at which gate will trip
Test 5 Trip Voltage	DC Supply = 19.5 VDC Turbine frequency equal to lower, upper and mid value of Test 4	Turbine Output Voltage	Determine minimum turbine output voltage at which gate will trip

NOTE: During Tests 4 and 5 the internal gate test points (Schmitt trigger, flip/flop, frequency selective network, and peak rectifier outputs) were measured and recorded and the waveforms were compared to standard test results.

Table 5-4

TEST CONDITIONS FOR ELECTRONIC GATE AND RELAY BOX

Environment	Levels	Test Conditions
Shock 8 millisecond duration 1/2 sine wave form	40, 60, 80, 100, 120, 140, 160 g's	One shock in each direction of three principle axes.
Random Vibration 1 minute with a spectrum of 0.08 g^2/cps at 20 to 400 cps and 0.2 g^2/cps at 400 to 2000 cps	18.7, 23.4, 28.1, 32.8, 37.5 g (RMS)	Vibration along each of three principle axes. Unit cycled in operating mode.
Temperature One-hour soak at each temperature	0, -20, -40, -60, -80° F +160, +200, +240, +280, +320, +360, +400° F	Special functional test run after soak at each temperature to evaluate performance.
Voltage	Decreasing in 1-volt steps from 20 VDC until unit fails to operate. Increasing in 4-volt steps from 30 VDC to 42 VDC. (Gate limited to 32 VDC maximum by internal Zeners)	Special functional test run at each voltage to evaluate performance.
Duty Cycle (Electronic Gate Only)	Five days at 160° F, 5 days at 0° F, 5 days at 160° F and final 5 days at 0° F	First 10 days in reset mode, last ten days in tripped mode. Functional Test run at end of each 5 days to evaluate performance.

Table 5-5

TEST CONDITIONS FOR ENGINE HARNESS

Environment	Levels	Test Conditions
Temperature One-hour soak at each temperature	+160, +200, +240, +280, +320, +360, +400° F	Special functional test for harness run at each level to evaluate performance.
Dielectric Strength	600, 1200, 1800, 2400 volts RMS	Applied for 5 seconds in accordance with Method 301 of MIL-STD-202.
Insulation Resistance	100 VDC insulation resistance test	Performed after each level of dielectric strength testing per Method 302 of MIL-STD-202.

Table 5-6

SUMMARY OF ELEVATED STRESS TEST RESULTS

Unit	Stress Conditions	Results
Electronic Gate	Shock	Passed all intended functional requirements, operated within specification, and remained operable throughout all levels.
	Temperature	Passed all intended functional requirements over the range of -80 to +280° F, operated within specification over the range of -40 to +200° F, and remained operable throughout all levels.
	Voltage	Passed all intended functional requirements over the range of 14 to 32 VDC, operated within specification over the range of 17 to 32 VDC, and remained operable throughout all levels.
	Duty Cycle	Passed all intended functional requirements, operated within specification, and remained operable over the 20 day span at 0° F and 160° F temperature limits used during test.
	Mounting and Structure	Satisfactory, no detrimental effects observed.
Relay Box	Shock	Passed all intended functional requirements, operated within specification, and remained operable throughout all levels.
	Temperature	Passed all intended functional requirements over the range of -80 to +360° F, operated within specification over the range of -80 to +320° F. Inoperable subsequent to test at 400° F.

Table 5-6 (Cont.)

Unit	Stress Conditions	Results
Relay Box (Cont.)	Voltage	Passed all intended functional requirements over the range of 12.8 to 42.5 VDC, operated within specification over the range of 12.8 to 42.5 VDC, and remained operable throughout all levels.
	Vibration	Passed all intended functional requirements, operated within specification, and remained operable throughout all levels.
	Mounting and Structure	Satisfactory – No detrimental effects observed.
Engine Harness	Temperature Insulation Resistance Dielectric Strength	Passed all intended functional requirements, operated within specification, and remained operable throughout all tests.

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5.4 TURBINE PUMP TEST PROGRAM

Tests were conducted to determine turbopump and electrical control hardware start transient operation and performance characteristics. These tests were under sea level conditions and included the YLR 81-BA-11 engine fuel and oxidizer valve configurations proposed for the modified XLR 81-BA-13 engine. The tests were also to provide a basis for predicting engine operating characteristics at sea level and at altitude. Specific objectives were as follows:

- Determine the effects of engine modification on the start system operating characteristics including the effect of the LMSC redesigned vehicle aft safe-arm J-box (ASA J-box).
- Determine the modified engine start transient characteristics over the required range of operating temperatures and start tank pressures.
- Verify acceptability of main propellant valve selection.
- Establish preliminary transient performance values for preflow and postflow, start tank recharge time, and predicted time to 75 percent rated thrust for the modified engine configuration.
- Determine and evaluate the modified engine operating and transient performance characteristics under various imposed malfunction conditions.
- Determine the pressure pulse damping effect of a snubbing orifice in the oxidizer feed pressure switch (OFPS) sensing line.
- Evaluate the OFPS mechanical durability.

The turbine pump assembly tests were conducted at the Bell Aerosystems Company, Rocket Test Facility, test cell Y-2, from 12 November 1965 to 9 March 1966.

5.4.1 Test Configuration

The turbine pump assembly used in the tests (S/N K3TA) was assembled from components obtained from the previous XLR 81-BA-13 engine program residual hardware. A production relay box detail design was not established at the time the turbine pump

tests were initiated and a preliminary design relay box with additional instrumentation circuitry and no EMI suppression across the relay was used as shown in Fig. 5-5. An oxidizer valve, pressure switches, and fuel valve assembly were obtained from YLR 81-BA-11 engine test and production hardware.

The main fuel and oxidizer valves were mounted externally, and as shown in Figs. 5-6 and 5-7, respectively, to simulate the normal engine assembly operating effects. The pump discharge to the oxidizer and fuel valves was circulated back to the main propellant tanks. An orifice was located downstream of the propellant valves to simulate the thrust chamber back pressure.

For the initial tests, a pressure switch simulating the oxidizer manifold pressure switch (OMPS) was installed between the oxidizer valve and the orifice used to simulate chamber back pressure. Because the large fill volume of the thrust chamber regenerative cooling passages was omitted from the system, the simulated OMPS actuation pressure switch was set higher than the design OMPS to more accurately simulate the XLR 81-BA-13 engine start transient time to OMPS actuation.

The turbine pump assembly components, except for the electronic gate and pressure switch relay box, were enclosed within an environmental chamber. Thus, the propellants in the main tanks could be recirculated for conditioning to the temperatures specified in Table 5-7.

Oxidizer Recirculation System. Initial test results demonstrated that the oxidizer recirculation system did not provide a valid simulation of the thrust chamber cooling passage transient flow-pressure characteristics. Consequently, a thrust chamber with an oxidizer valve and a flight configuration OMPS were incorporated into the pump discharge oxidizer system to provide an alternate oxidizer flow circuit while the fuel system remained as originally installed. A two-second operational limit using the thrust chamber was imposed to minimize the oxidizer flow through the thrust chamber and into the test cell. The alternate configuration providing simulated start transient flow characteristics during short duration runs with the thrust chamber and oxidizer recirculation capability for long duration runs is shown schematically in Fig. 5-6.

After the thrust chamber installation, the results from tests 2041 through 2047, with an OFPS set at 235 psi actuation pressure, indicated premature switch actuation with the XLR 81-BA-13 engine start transient characteristics. Consequently, effective at test 2048, an OFPS with an actuation pressure setting of 391 (±24) psia was incorporated.

Electrical Control Circuit. Test 2048 start sequence, simulating an oxidizer manifold pressure switch (OMPS) failure to actuate (OFPS commanding), was terminated due to an electronic gate shutdown. Investigation of the data indicated no turbine overspeed condition existed. It was concluded that the shutdown condition occurred because the electrical load switched by the OFPS was conducted through the gate via the main power line for the pilot operated solenoid valve (POSV) and pressure switches as shown in Fig. 5-5. This was a fluctuating inductive load and caused the gate to trip. The electronic gate S/N 7 was replaced and EMI suppression was added across the lock-in relay in the pressure switch relay box. This modification rectified the situation and the two OFPS commanding tests were completed successfully. The electronic gate S/N 7 was then checked functionally and found to be acceptable. Consequently, another test was conducted under the original conditions (no EMI suppression in the pressure switch relay box and using gate S/N 7) and test results were similar to the electronic gate shutdown during the start transient OFPS chatter. The EMI circuit was reinstalled across the lock-in relay and the test was repeated. The gate again tripped, indicating that the tripping was caused by electrical load fluctuation generated by the OFPS chatter rather than by a malfunction within the gate. The electronic gate S/N 7 was replaced with another gate for the remaining turbine pump tests which eliminated the premature shutdown condition during subsequent testing. The EMI suppression circuit remained across the lock-in relay for the remaining tests.

Test Cell Electrical Control Circuit. The turbine pump test cell electrical control circuit was modified on 27 December 1965 to determine the effects, if any, of the LMSC aft safe/arm junction box (ASA J-box) on the XLR 81-BA-13 modified engine start and shutdown transients. The main power to the engine solenoid valves was

routed directly from the simulated ASA J-box to the engine gas generator solenoid valves and the pressure switch relay box as shown in Fig. 5-8. Thus, the high current load was removed from the electronic gate relays which existed with the previously proposed configuration shown in Fig. 5-5. This new configuration, equivalent to vehicle use, eliminated the fluctuating inductive load currents to the POSV through the electronic gate circuit under conditions of OFPS chatter.

Low Suction Pressure Tests. Turbine pump test cell modifications were initiated on 3 January 1966 to incorporate low suction pressure test capability. To perform the low suction pressure test runs, it was necessary to install an auxiliary main propellant tank system due to an excessive pressure drop in the standard test cell propellant feed system. In addition, the alternate propellant recirculation system was used in the oxidizer system to facilitate diverting the oxidizer pump discharge flow through the thrust chamber for bit impulse runs or recirculating the flow to the propellant tank for long duration runs. This arrangement is shown in Fig. 5-6. Modifications to the propellant system were incorporated to resolve the discrepancies noted in Table 5-7 for tests 2080 through 2085. However, prior to completing the required modifications, the low suction pressure tests were cancelled to expedite OFPS durability testing, and the test cell was converted back to the test cell main propellant feed system configuration.

Pressure Switch Junction Box. A turbine pump test cell electrical system modification was accomplished on 25 February 1966 to incorporate the pressure switch junction box in lieu of the pressure switch relay box to coincide with changed engine design configuration (Ref: paragraph 5.2). Additional tests were conducted to determine the effects on the fuel valve assembly operation with the OFPS commanding (OMPS disabled) and the pressure switch junction box replacing the relay box.

Instrumentation. The turbine pump assembly was instrumented as shown schematically in Figs. 5-5, 5-6, 5-8, and 5-9. Oxidizer feed and manifold pressures (OFP and OMP) were monitored with close-coupled transducers as shown in Fig. 5-6.

5.4.2 Test Results

Table 5-7 shows the average values of time to thrust chamber ignition, turbine speed at thrust chamber ignition, and oxidizer preflow and postflow for the two main propellant valve configurations (YLR 81-BA-11 and XLR 81-BA-13) considered for the modified XLR 81-BA-13 engine. A comparison of these values indicated that the start transients for the two configurations were similar and that engine level tests could begin with confidence that the required oxidizer preflow would be achieved. The time to ignition is slightly less for the modified XLR 81-BA-13 engine (relative to the YLR 81-BA-11 engine); the turbine speed at ignition is nearly the same, indicating a higher turbine acceleration for the XLR 81-BA-13 engine.

Start Tank Pressure and Hardware and Propellant Temperature Effects. The effects on time to thrust chamber ignition and the turbine speed at thrust chamber ignition are shown in Table 5-8 (tests 2040 through 2068). Lower start tank pressures and temperatures result in longer times to ignition and lower turbine speeds at ignition due to the lower propellant flow rate to the gas generator and lower generator performance at reduced temperature. These conditions reduced the available energy for turbine acceleration.

A significant effect of start tank initial conditions on time to thrust chamber ignition is shown in Table 5-9 for the malfunction tests 2112, 2113, and 2114 which simulated the extreme condition of complete loss of propellant from either and both start tanks. The results show successful engine start and start tank recharge at +60°F temperature and 40 psig pump suction pressure. However, the time to thrust chamber ignition increased primarily as a function of oxidizer start tank propellant loss. Loss of fuel start tank propellant produced a time to thrust chamber ignition of 3 seconds, however loss of oxidizer start tank propellant resulted in a time to thrust chamber ignition of 12 seconds. Variable fuel start tank propellant had relatively minor effect on time to thrust chamber ignition for the condition of no oxidizer in the oxidizer start tank. The predicted thrust chamber ignition occurred approximately coincident with

attainment of 75 percent turbine speed with an empty fuel start tank and occurred slightly prior to 75 percent turbine speed with an empty oxidizer start tank or both start tanks empty. Regarding application of this test data to flight, it appears that if fuel start tank propellant is lost in flight, the chances of engine restart are reasonably good. However, for the case of lost propellant from the oxidizer start tank, significant delay in start transient events was observed and high pump suction pressures were utilized during these tests compared with lower predicted flight suction pressures. Therefore the limited test data obtained for these conditions presently prohibits an accurate quantitative prediction of flight start transient characteristics with lost oxidizer start tank propellant.

Pressure switch malfunction conditions, i.e. actuated at start signal (fuel lead start sequence) and OMPS failure to actuate (OFPS commanding start sequence) also affect time to thrust chamber ignition and turbine speed at ignition, as discussed later in this section.

Oxidizer Preflow. The nominal oxidizer preflow values obtained at sea level during the turbine pump program agreed with the initial predicted preflow for the modified XLR 81-BA-13 engine and with the oxidizer preflow values quoted for the YLR 81-BA-11 production engine as shown in Table 5-7. The test results did not show a significant effect of main propellant valve configuration, turbine pump suction pressures, or initial start tank pressures (within the required operating limits) on oxidizer preflow.

OFPS Actuation Effects. The data obtained from tests 2041 through 2047 using an OFPS set at 235 psia actuation pressure showed that the switch actuated at two distinct intervals prior to normal OMPS actuation. The first actuation of the OFPS was the result of the pressure increase required to rupture the oxidizer frangible disc. The second actuation occurred when the combined effects of the increasing turbine speed and fluid flow resistance in the thrust chamber cooling passages resulted in an increasing pressure at the OFPS sufficient to actuate the switch. The premature

OFPS actuation was of sufficient duration to actuate and electrically lock-in the pressure switch relay box relay and energize the pilot operated solenoid valve (POSV). These premature OFPS actuations would result in an estimated 2.0 pound fuel lead and simultaneous injection for the first and second premature actuations, respectively, with the relay box electrical control system. Consequently, an OFPS with a higher actuation pressure was required to ensure an oxidizer lead start sequence. The YLR 81-BA-11 engine thrust chamber pressure switch was selected since it was identical in configuration to the initially selected OFPS except for the heavier column spring and correspondingly higher nominal actuation pressure setting of approximately 390 psig.

The tests conducted to evaluate the start transient characteristics with the OFPS actuation pressure of 390 psi revealed that the switch did not actuate and lock-in on the first actuation. Instead, the switch chattered (oscillated between the actuate and deactuate positions) due to the oxidizer pump pressure perturbations which were greater than the pressure switch actuation/deactuation pressure settings. This chattering condition would continue until the oxidizer system minimum pressure increased to a value greater than the switch deactuation pressure setting. This pressure increase occurred as a result of thrust chamber ignition chamber pressure rise or turbine speed increase resulting in the increasing pump discharge pressure. During a normal start sequence, the OFPS chatter would result in 5 inductive load carrying cycles (POSV coil) prior to OMPS actuation and 25 noninductive load carrying cycles between OMPS actuation and thrust chamber pressure rise. During an OFPS commanding start sequence (simulating OMPS failure to actuate) with the OFPS at nominal actuation pressure setting, a total of 30 inductive load carrying cycles would occur prior to thrust chamber ignition.

This high frequency OFPS chatter operation during a normal start transient required the additional switch durability tests (tests 2093 through 2101) discussed in paragraph 5.7 and an evaluation of an OFPS pressure sensing tube assembly orifice for pressure perturbation attenuation capability (tests 2090 through 2092). The addition of different sized orifices in the OFPS sensing line was investigated to determine if oxidizer feed pressure pulsing could be attenuated. The test results indicated that an orifice with

less than 0.10 inch diameter would be necessary to provide the required pressure pulse attenuation at the switch. An orifice of this size would cause excessive delay in OFPS actuation time. Consequently, the orificed sensing tube was not recommended to attenuate OFPS chatter.

The use of the OFPS with a nominal actuation pressure of 390 psi to command engine start resulted in the OFPS functioning as a backup command switch and an increase in oxidizer preflow in the event of an OMPS failure to actuate, as shown in Table 5-10. The turbine pump operating characteristics have a greater affect on the engine start transient parameters with the OFPS controlled start sequence than with OMPS in command. The turbine acceleration rate affects the time to achieve OFPS actuation pressure and therefore the time to energize the POSV. Consequently, variations in temperature, initial start tank pressures (which affect turbine acceleration), and OFPS actuation pressure setting (affecting the required turbine speed), in addition to the POSV actuation characteristics with an intermittent (OFPS chatter) signal would be predicted to significantly affect start transient parameters. Therefore, turbine pump test (test 2069) was conducted at the operating limits of low start tank pressures (600 psia), low hardware and propellant temperature (+10°F), and the maximum possible OFPS actuation pressure of 414 psig to determine the maximum effect on the start transient parameters. The actual value of preflow for this condition was not determined, since the OFPS never actuated the POSV during the 2-second duration run.

The possibility of a fuel lead, due to the OFPS having the minimum possible actuation pressure of 368 psig, high start tank pressures (1400 psia), and high hardware and propellant temperatures (+140°F) was also investigated (test 2070). The OFPS actuated at 0.918 seconds after start signal which was 0.091 seconds after the OMPS actuation which was monitored for comparison with the OFPS actuation time. Consequently, an oxidizer lead of 5 pounds resulted with the pressure switch relay box control circuitry shown in Fig. 5-8. Since the control circuitry lock-in relay could affect start transient parameters, these tests were repeated (tests 2104, 2107, and 2105) incorporating the pressure switch junction box design change, as shown in Fig. 5-9.

The junction box test results are compared with the aforementioned relay box test results in Table 5-10 and show no significant differences in the start transient parameters, however, tne following condition was observed which resulted from the removal of the lock-in relay.

Examination of the data indicated that the fuel valve repeatedly shuttled between the fully opened and closed positions three times during the 2-second duration run. The fuel valve shuttling is explained as follows. The pressure at the OFPS increased during the start transient until the switch was actuated, thereby commanding the POSV to open the main fuel valve. As soon as the fuel valve opened, the fuel flow placed an additional load on the pump and hence on the turbine, causing a decrease in turbine speed. The decrease was sufficient to reduce the mean pressure at the OFPS causing the switch to deactuate, remove electrical power from the POSV, and thereby close the fuel valve. This cycle operation would repeat; however, this situation was determined to be peculiar to the turbine pump level operation which does not allow simulation of the effects of thrust chamber pressure which rises after the fuel valve opens. It was predicted that on the engine level, once the fuel valve opened and permitted flow into the thrust chamber, the chamber pressure would rise rapidly due to combustion. Rise in chamber pressure would produce a corresponding propellant feed system pressure rise to a value above the OFPS actuation pressure setting, which in turn would provide positive lock-in of the switch. This positive operation was verified during AEDC tests as discussed in Section 7.

Oxidizer Postflow. An evaluation of the turbine pump test results was performed to determine the effects of the engine modifications on oxidizer postflow. Results showed an oxidizer postflow increase from 18.5 pounds with the former XLR 81-BA-13 engine main oxidizer valve configuration to about 28.8 pounds with the proposed main oxidizer valve from the XLR 81-BA-11 engine at 24 psia pump suction pressures. However, consistent with the criteria to modify the engine as closely as possible to the YLR 81-BA-11 engine thrust chamber start sequence configuration, the YLR 81-BA-11

engine configuration main valves were retained. The turbine pump effects on oxidizer postflow have been determined to be primarily a function of the turbine speed and oxidizer pump suction pressure present at shutdown.

Oxidizer postflow will continue until turbine speed delays sufficiently to allow the oxidizer valve to close under spring load. The engine main fuel valve closes prior to the gas generator solenoid valve closing which terminates propellant flow to the gas generator. Turbine speed will increase when the main fuel valve closes unloading the fuel pump, until the gas generator operation is terminated. This turbine speed increase with the corresponding effect on oxidizer postflow is primarily a function of the oxidizer gas generator solenoid valve closing time. This valve closes in less time than the corresponding YLR 81-BA-11 engine bi-propellant valve, and results in a lower turbine speed increase at shutdown and a correspondingly lower oxidizer postflow of 28.8 pounds compared to 31.2 pounds for the YLR 81-BA-11 engine.

A simulated aft safe arm junction box (ASA J-box) was incorporated into the turbine pump electrical circuitry to evaluate the possible effects of the engine electrical design change on the gas generator valve closing times and oxidizer postflow. The significant results of these tests compared with data obtained with the gas generator solenoid valve electrical power supply routed through the electronic gate are presented in Table 5-11. The comparison shows that the mean closing times of the gas generator fuel and oxidizer solenoid valves and the main fuel valve were not altered by the electrical control circuit design change. Shutdown turbine speed increase and nominal oxidizer postflow are therefore not affected by the ASA J-box.

Malfunction Tests. Malfunction tests were conducted to determine the effects of fuel and oxidizer depletion at start and at steady state, and to evaluate the effects of the ASA J-box circuitry on an electronic gate turbine overspeed shutdown. The significant results are:

a. Turbine overspeed shutdown was not achieved for the condition of fuel depletion either at start or during operation. The turbine speed reached a peak speed of 28,000 rpm when depletion occurred during steady state operation and the run was terminated when the fuel start tank was depleted.

b. For oxidizer depletion at start or during steady state operation, the electronic gate initiated turbine overspeed shutdown at 29,500 rpm. The shutdown control circuitry and hardware response times resulted in a maximum turbine speed of 32,000 rpm for depletion either at start or during steady state operation.

5.4.3 Conclusions

The turbine pump tests demonstrated reliable and predictable operation of the electrical control and hardware modifications over the range of operating conditions imposed at sea level pressure. A basis for predicting engine level operating characteristics at both sea level and altitude was provided.

Values of the start transient parameters were derived which agreed satisfactorily with initially predicted values. Start tank recharge time remained unchanged. Time to thrust chamber ignition was slightly less than the predicted value of 1.16 seconds. The oxidizer preflow of 6.0 pounds agreed well with the YLR 81-BA-11 engine value, and provided confidence that the formal engine level tests could proceed without further design change.

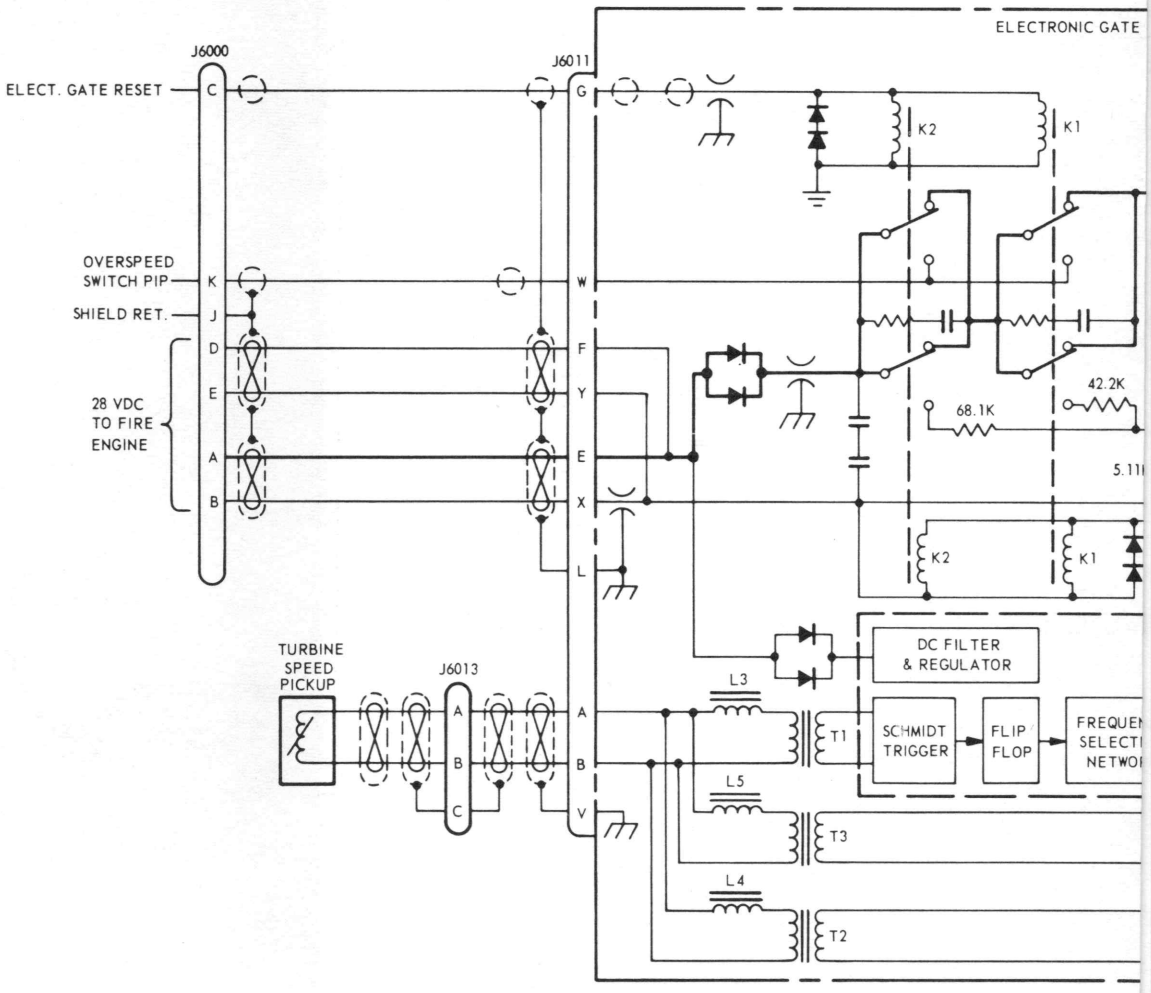
The pump tests showed no significant difference in start transient operation when either the YLR 81-BA-11 or XLR 81-BA-13 engine valves were used in conjunction with the modified electrical control circuitry. However, the YLR 81-BA-11 engine valves were selected for incorporation into the modified engine configuration, primarily based on the philosophy of changing as much as possible to the YLR 81-BA-11 engine configuration. With the YLR 81-BA-11 engine oxidizer valve, an oxidizer postflow of 29 pounds nominal is achieved, which is lower than the 31 pounds obtained with the YLR 81-BA-11 engine, but considerable higher than that associated with original XLR 81-BA-13 engine.

Malfunction tests on loss of propellant from either start tank demonstrated that the time to ignition is delayed by two seconds when fuel start tank propellant is lost, and by 11 seconds when oxidizer is lost from its start tank. Before these results can be applied to flight conditions, further analysis and perhaps testing would be required. Results of other malfunction tests were as predicted; in most instances results were obtained which were similar to those achieved during the original XLR 81-BA-13 engine development.

Reliable operation with the vehicle aft safe arm junction box in the electrical control circuit was verified.

Premature activation of the OFPS during early tests resulted in selection of a switch with a higher setting of 391 psig to guarantee an oxidizer lead in the event of an OMPS malfunction. With this setting, significant OFPS chatter occurred due to oxidizer pump discharge pressure perturbations during start transient pressure buildup. Therefore,tests were run to determine if an orifice in the OFPS sensing tube assembly would attenuate the pressure perturbations and eliminate OFPS chatter. Results indicated that the small orifice size required was not compatible with engine transient response characteristics. OFPS durability tests were then conducted which verified that the OFPS would operate properly after multicycling in a load carrying condition.

Significant effects of environment, pressure, malfunction and test conditions on transient performance and operation were obtained during the pump tests, thereby minimizing engine level test requirements. Confidence in the modified engine detailed design and configuration was established prior to initiating the formal engine flightworthiness demonstration tests at sea level and AEDC.


ELECTRONIC GATE
J6000
J6011
ELECT. GATE RESET
C
G
K2
K1
OVERSPEED
SWITCH PIP
K
W
SHIELD RET.
J
D
F
E
Y
28 VDC
TO FIRE
ENGINE
A
E
B
X
L
42.2K
68.1K
5.11
K2
K1
TURBINE
SPEED
PICKUP
J6013
DC FILTER
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V
T3
L4
T2

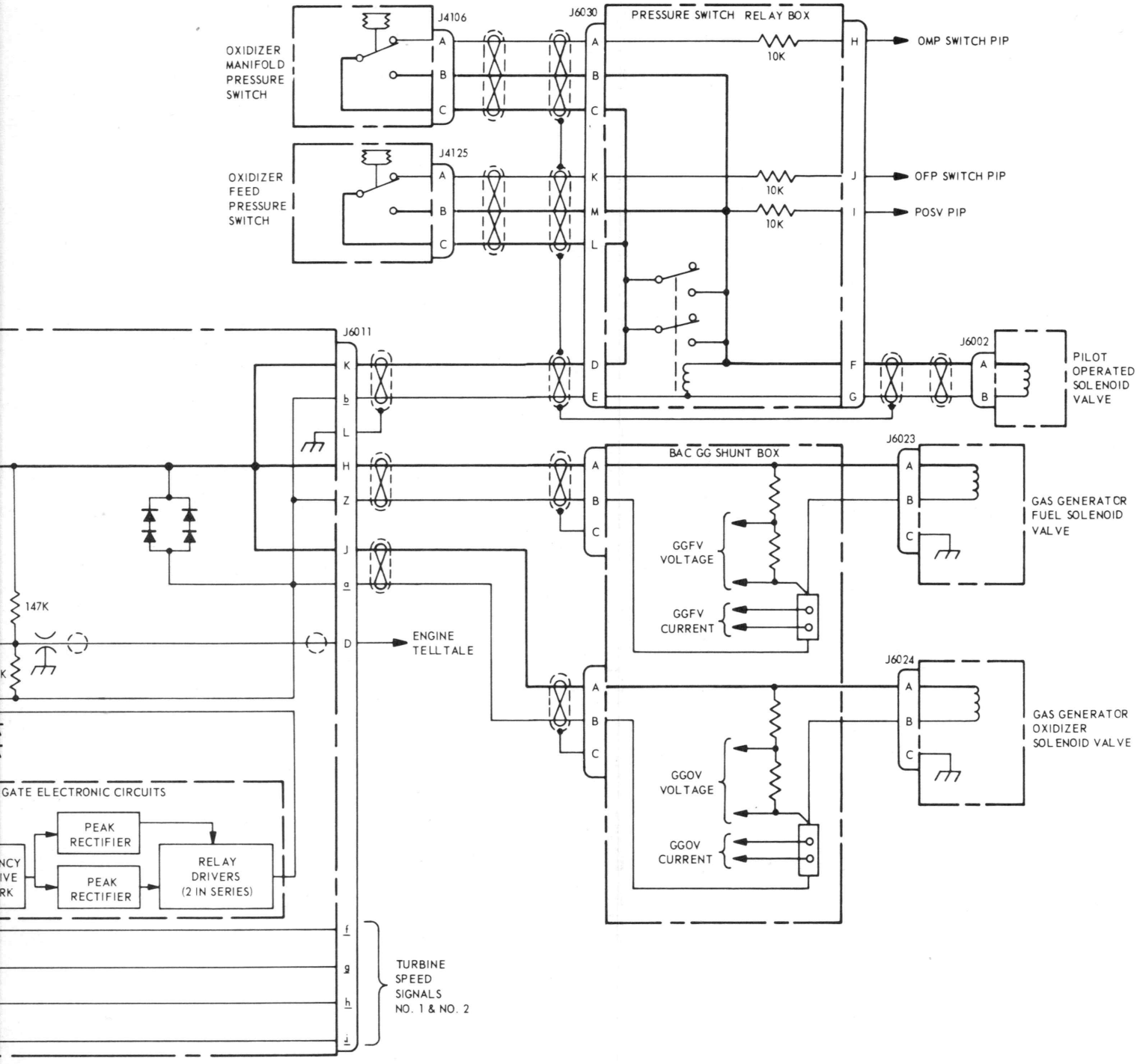
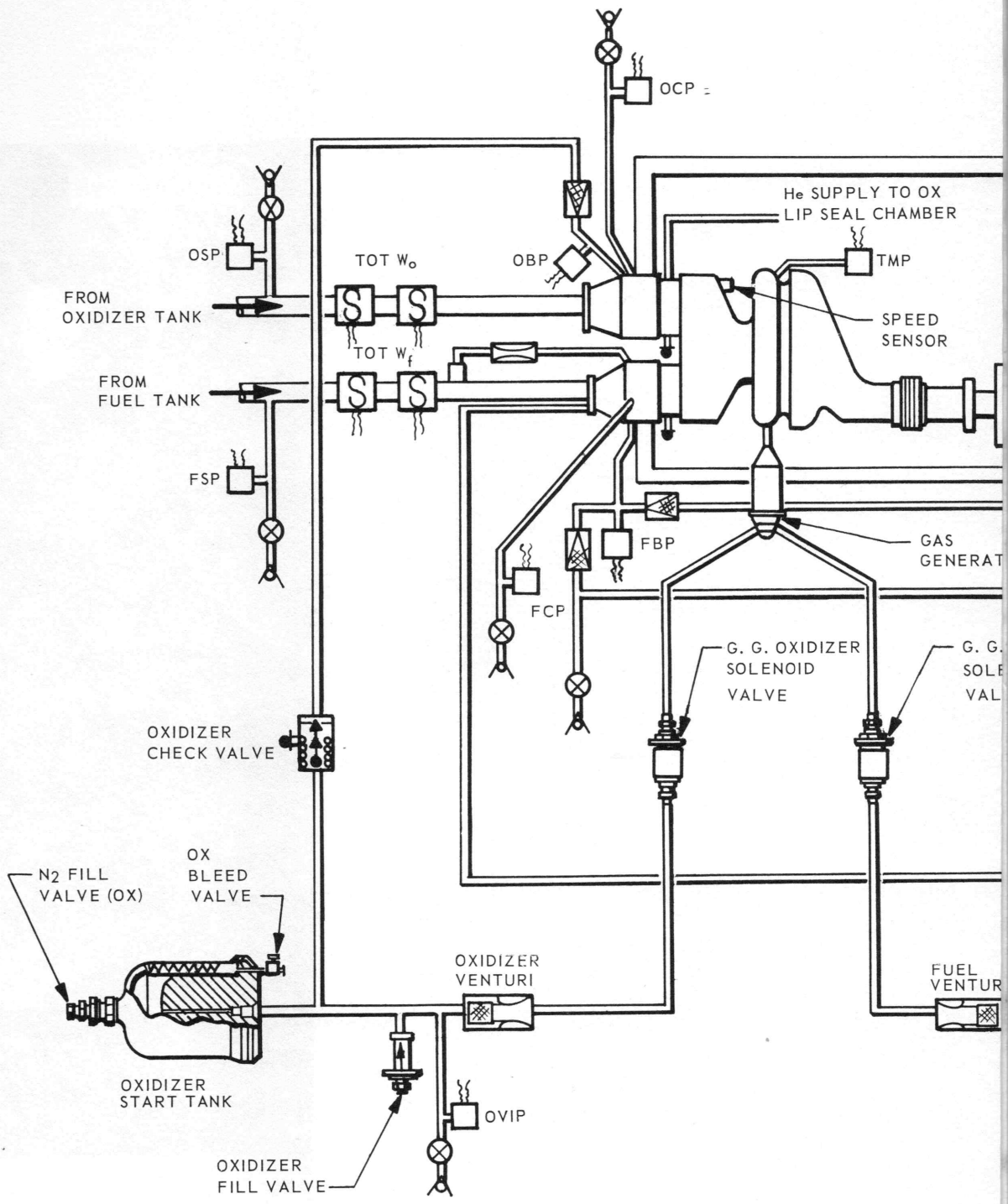


Fig. 5-5 Initial Turbine Pump Electrical Schematic


OCP
He SUPPLY TO OX
LIP SEAL CHAMBER
TMP
OSP
TOT W_o
OBP
FROM
OXIDIZER TANK
SPEED
SENSOR
TOT W_f
FROM
FUEL TANK
FSP
FBP
GAS
GENERAT
FCP
G. G. OXIDIZER
SOLENOID
VALVE
G. G.
SOLE
VAL
OXIDIZER
CHECK VALVE
N_2 FILL
VALVE (OX)
OX
BLEED
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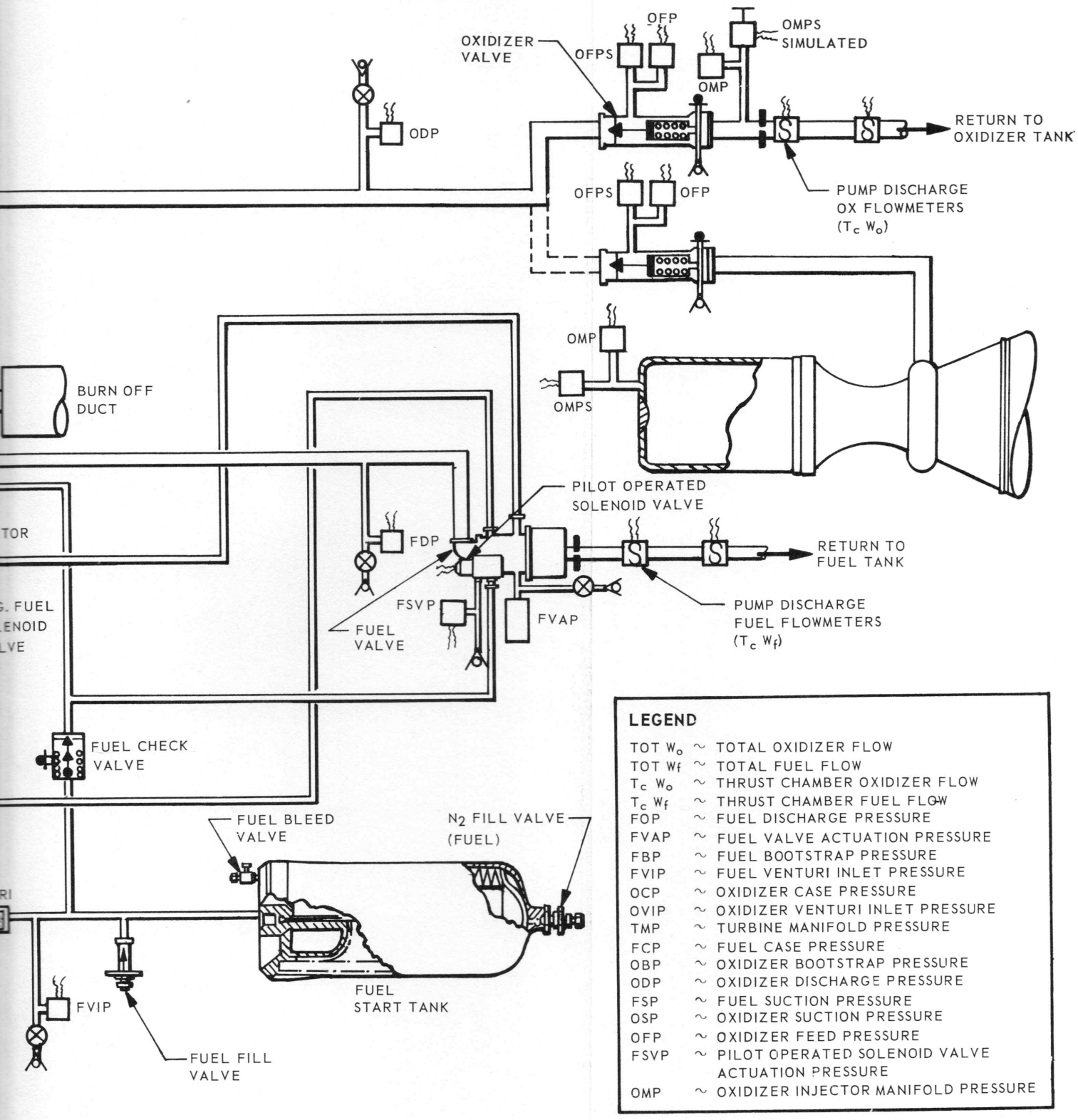
OXIDIZER
VENTURI
FUEL
VENTUR
OXIDIZER
START TANK
OVIP
OXIDIZER
FILL VALVE



LEGEND

TOT W_o	~	TOTAL OXIDIZER FLOW
TOT W_f	~	TOTAL FUEL FLOW
T_c W_o	~	THRUST CHAMBER OXIDIZER FLOW
T_c W_f	~	THRUST CHAMBER FUEL FLOW
FOP	~	FUEL DISCHARGE PRESSURE
FVAP	~	FUEL VALVE ACTUATION PRESSURE
FBP	~	FUEL BOOTSTRAP PRESSURE
FVIP	~	FUEL VENTURI INLET PRESSURE
OCP	~	OXIDIZER CASE PRESSURE
OVIP	~	OXIDIZER VENTURI INLET PRESSURE
TMP	~	TURBINE MANIFOLD PRESSURE
FCP	~	FUEL CASE PRESSURE
OBP	~	OXIDIZER BOOTSTRAP PRESSURE
ODP	~	OXIDIZER DISCHARGE PRESSURE
FSP	~	FUEL SUCTION PRESSURE
OSP	~	OXIDIZER SUCTION PRESSURE
OFP	~	OXIDIZER FEED PRESSURE
FSVP	~	PILOT OPERATED SOLENOID VALVE ACTUATION PRESSURE
OMP	~	OXIDIZER INJECTOR MANIFOLD PRESSURE

Fig. 5-6 Turbine Pump Hardware Schematic and Instrumentation Locations

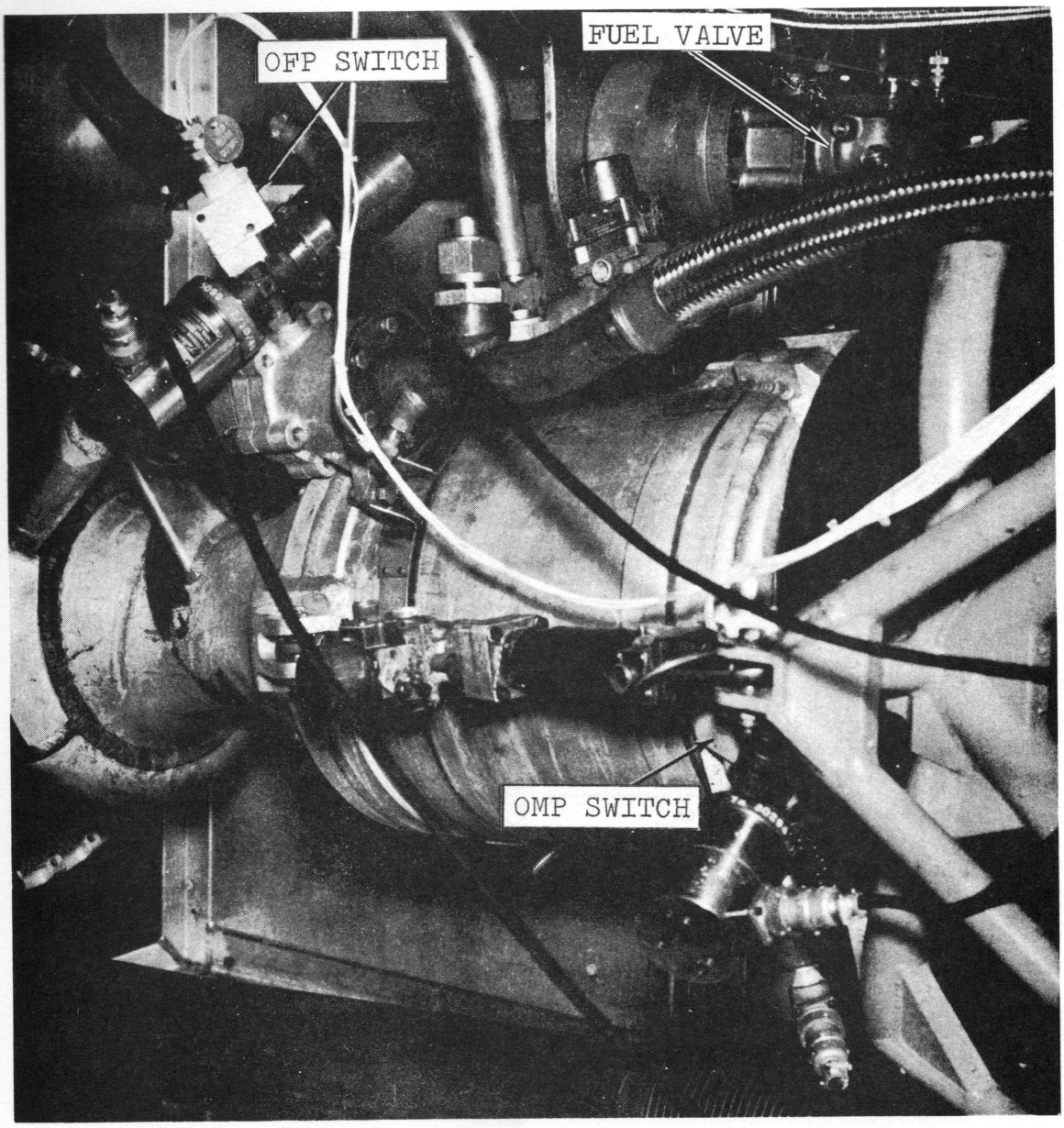


Fig. 5-7 Test Installation – Hardware Location

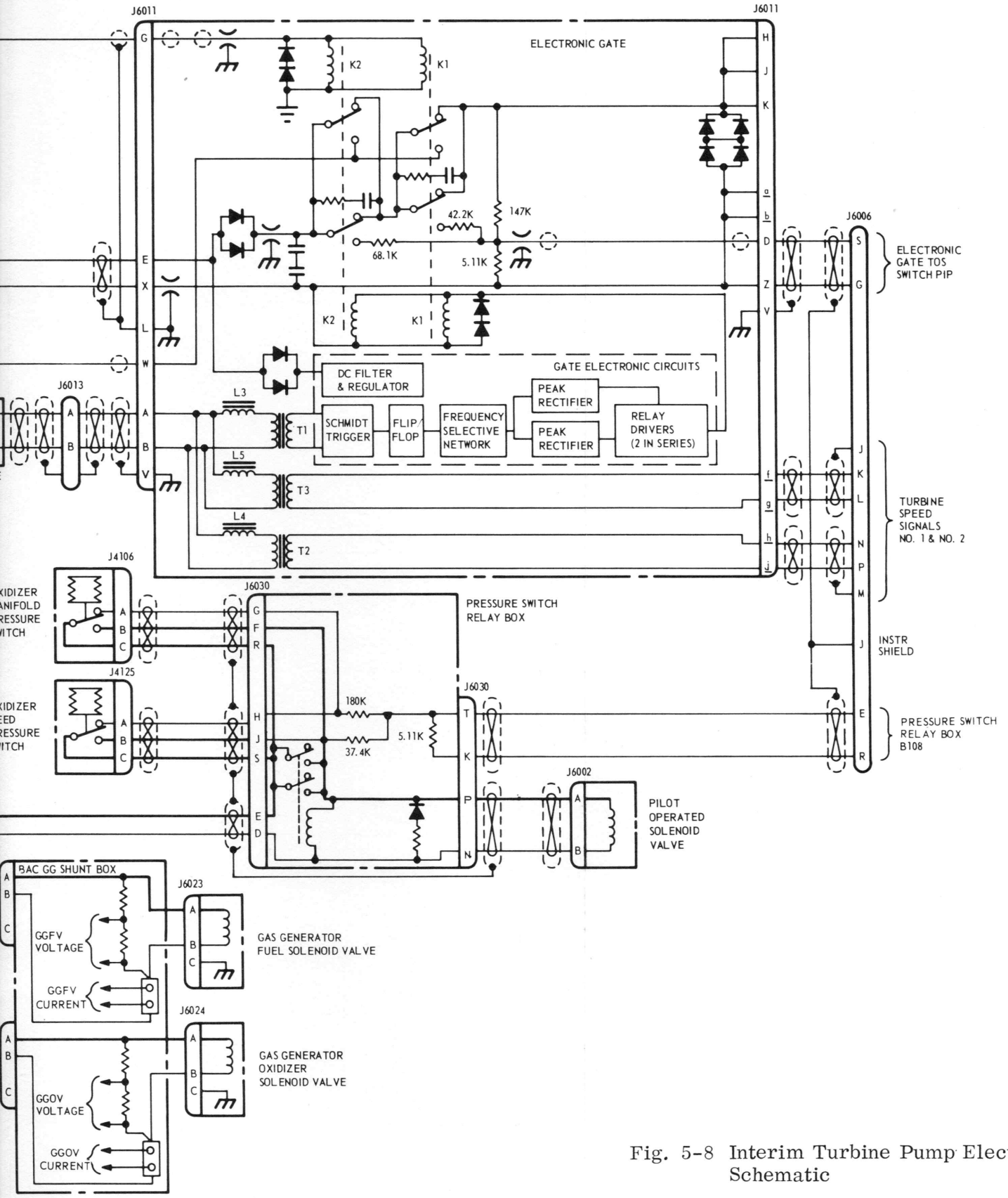
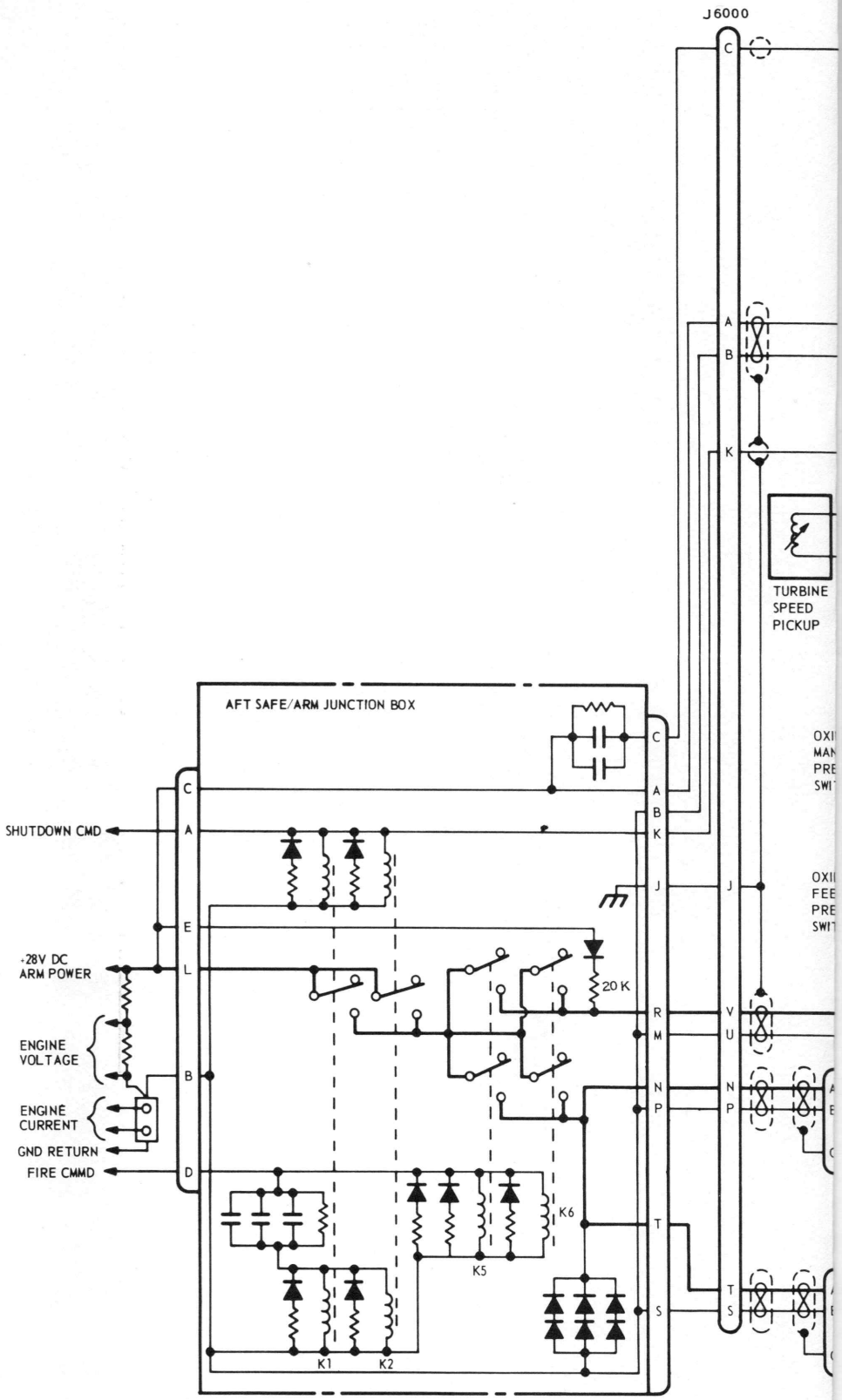


Fig. 5-8 Interim Turbine Pump Electrical Schematic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C
A
B
K
J
TURBINE
SPEED
PICKUP
AFT SAFE/ARM JUNCTION BOX
C
A
B
K
J
SHUTDOWN CMD
E
+28V DC
ARM POWER
L
20 K
R
V
M
U
ENGINE
VOLTAGE
B
N
P
ENGINE
CURRENT
GND RETURN
FIRE CMMD
D
K6
K5
T
S
K1
K2

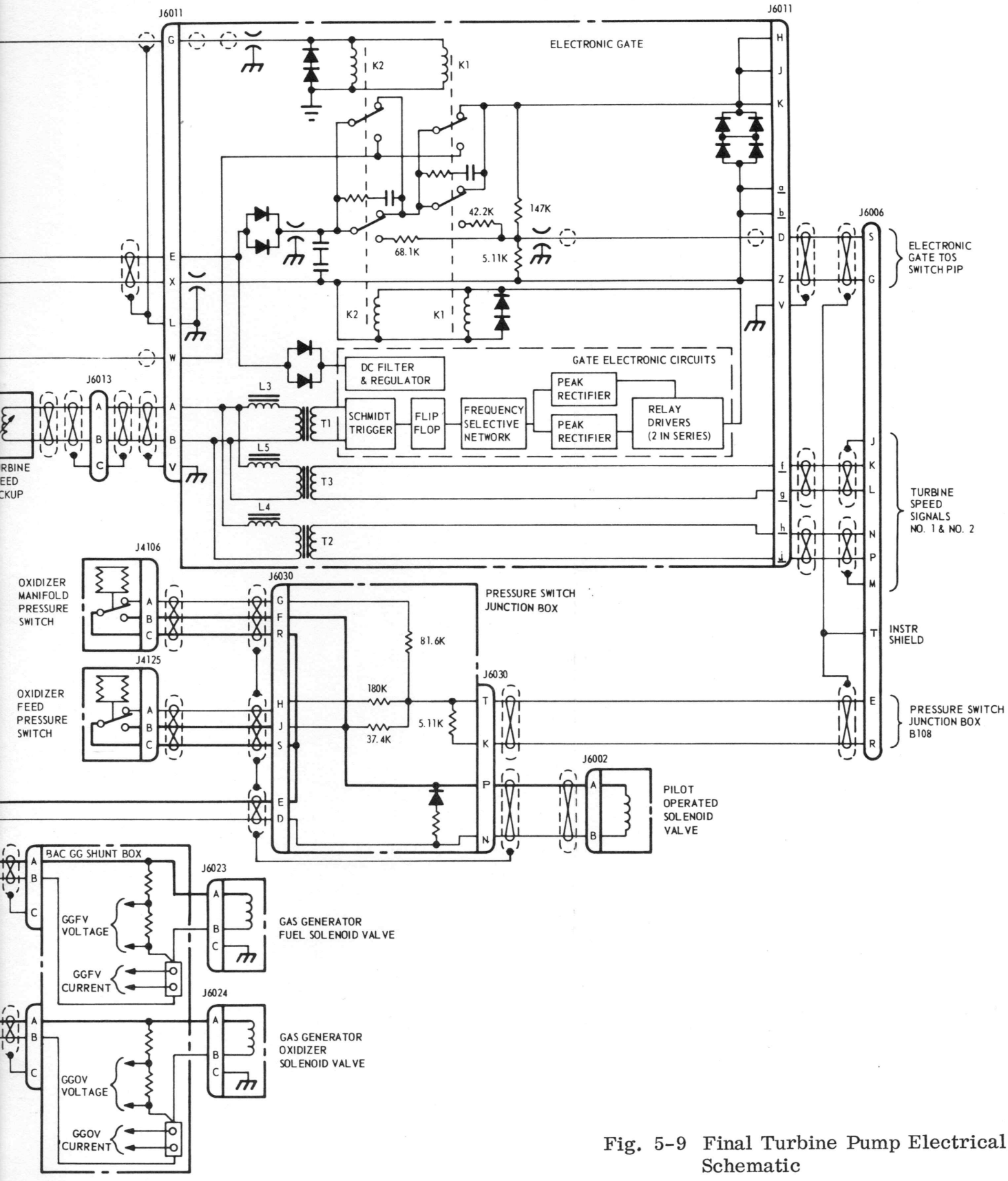


Fig. 5-9 Final Turbine Pump Electrical Schematic

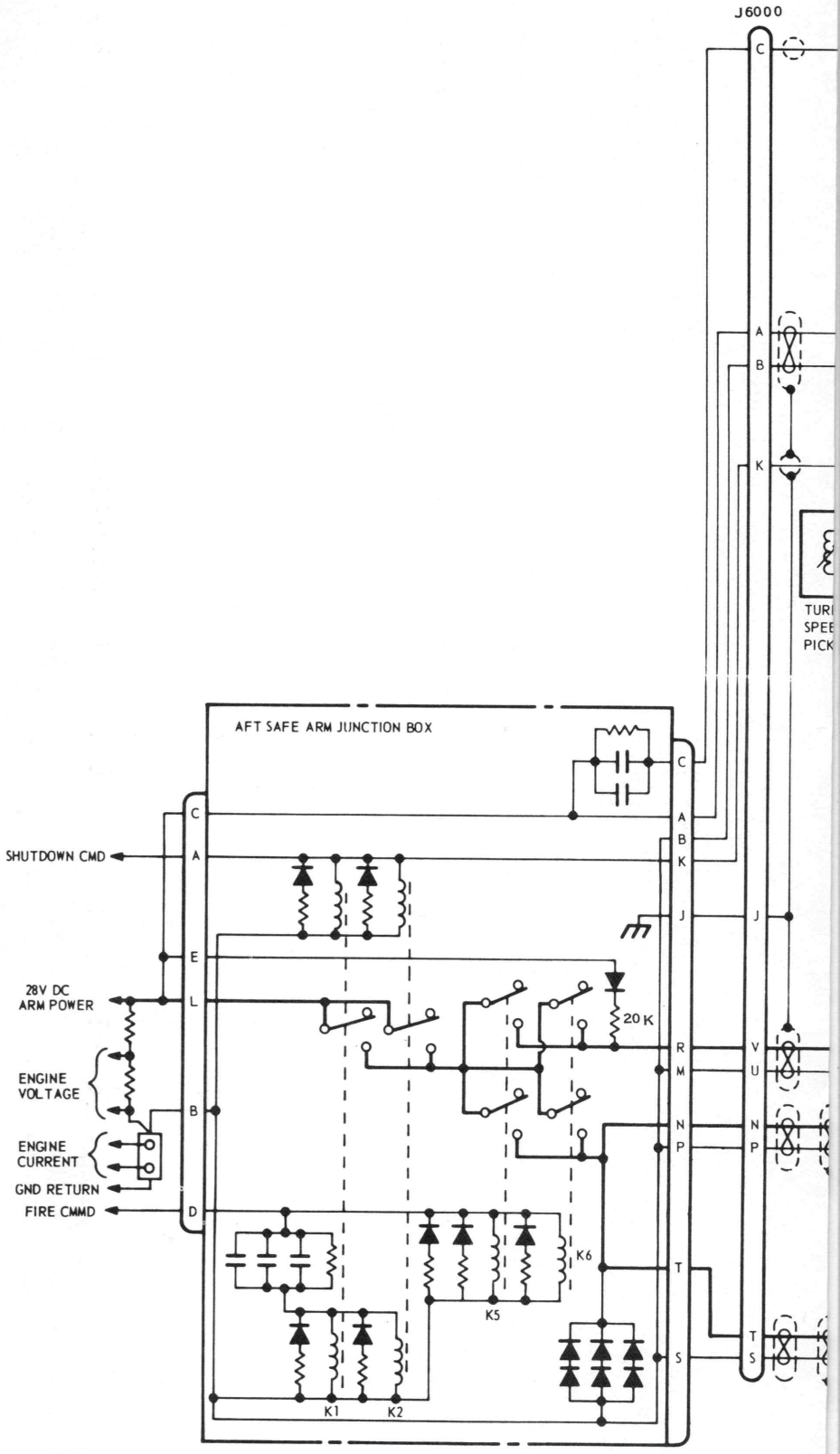

J6000
C
A
B
K
J
AFT SAFE ARM JUNCTION BOX
SHUTDOWN CMD
28V DC
ARM POWER
ENGINE
VOLTAGE
ENGINE
CURRENT
GND RETURN
FIRE CMMD
E
L
B
D
20 K
K6
K5
K1
K2
R
M
N
P
T
S
V
U

Table 5-7

COMPARISON OF NOMINAL TRANSIENT PERFORMANCE PARAMETERS

Test Article	Time to Thrust Chamber Ignition (sec)	Turbine Speed at Thrust Chamber Ignition (rpm)	Propellant Preflow (Oxidizer) (lb)	Propellant Postflow (Oxidizer) (lb)
Modified XLR 81-BA-13 Pump with YLR 81-BA-11 Main Fuel and Oxidizer Valves	1.042	22,250	6.0	28.0
YLR 81-BA-11 Production Engine	1.16 ±0.12	22,290 ±1368	5.3 ±2.2	33.5 ±3.7

Note: 1. YLR 81-BA-11 deviations are family 3σ values.

Test Conditions: Temperature (Hardware and Propellants) = 60° F
Altitude = Sea Level
Start Tank Pressure = 1000 psia
Fuel Suction Pressure = 24 psia
Oxidizer Suction Pressure = 24 psia

Table 5-8

TURBINE PUMP TEST PLAN AND TEST RESULTS

Series	Purpose	Test Y-2	Date	Test Config.	Duration (sec)	Hdwe. & Prop. Temp. (°F)	Start Tank Press. (psia)	Time to TC Ignition (sec)	Prop. Preflow (lb)	Turbine Speed at TC Ign. (rpm)	Oxidizer Postflow (lb)	Results
1	Calibrate turbine pump after buildup	2040A	11-12-65	1	60	60	1000	0.89	0.4 (F)	20,050	30.8	Unit satisfied all operational spec. requirements. Start characteristics similar to XLR 81-BA-13 (fuel lead)
		B		1	2	*	**	0.80	1.5 (F)	21,700	31.2	
2	Investigate the effects of varying start tank loadings on start transients using OMP-actuated POSV and YLR 81-BA-11 Prop. valves	2041	11-17-65	2	2	60	1000	1.10	7.4	22,100	27.8	OFPS (235 psig) momentarily actuated by pressure spike (oxid disc rupture) at approximately 0.5 sec. Probable fuel lead if OFPS had been in command of POSV.
		2042	11-18-65	2	2	60	1000	1.09	7.1	22,600	29.3	
		2043	11-19-65	2	2	60	675	1.13	6.1	20,760	28.0	
		2044	11-19-65	2	2	60	1300	0.97	6.2	23,820	29.5	
3	Investigate the effects of varying start tank loadings on start transients using OMP-actuated POSV and XLR 81-BA-13 Prop. valves	2045A	11-20-65	3	2	60	1000	1.07	6.6	23,000	18.6	Results similar to tests 2041-2044. Notes 1 and 2 apply.
		B		3	2	*	**	1.06	5.4	22,670	18.6	
		2046A	11-21-65	3	2	60	675	1.20	5.8	21,160	17.8	
		B		3	2	*	**	1.12	5.3	21,680	18.3	
		2047A	11-21-65	3	2	60	1300	0.98	5.5	23,970	18.8	
		B		3	2	*	**	1.07	5.2	22,520	18.5	
4	Determine turbine pump start transient characteristics using the OFPS (380 psig make) to actuate the POSV	2048A	11-21-65	4	1.15	60	1000	–	–	–	35.7	Malfunction shutdown due to gate sensitivity to switch chatter. Replaced electronic gate and added EMI suppression for test 2049. Notes 1 and 2 apply.
		2049A	11-23-65	4	2	60	1000	1.27	15.2	24,670	29.1	
		B		4	2	60	**	1.52	22.8	24,260	29.9	
5	Investigate the effects of varying start tank loadings on turbine pump start/shutdown transients with hardware and propellant temperature variation	2050A	11-24-65	5	2	60	1000	1.05	5.8	22,110	27.8	Oxidizer preflows (6.0 lbs) and oxidizer postflows (28.0 lbs) were found to be insensitive to start tank loadings and propellant temperature variation. Test 2056B was terminated due to insufficient propellant temperature in phase A (125°F). Note 1 and 2 apply.
		B		5	2	*	**	1.09	5.7	21,470	26.7	
		2051A	11-25-65	5	2	60	800	1.10	5.8	21,340	28.0	
		B		5	2	*	**	1.08	5.2	21,470	28.1	
		2052A	11-26-65	5	2	10	1000	1.07	6.2	22,200	27.8	
		B		5	2	*	**	1.07	6.2	22,208	28.0	
		2053A	11-27-65	5	2	10	800	1.17	6.9	21,112	27.0	
		B		5	2	*	**	1.10	5.4	21,293	27.7	
		2054A	11-28-65	5	2	10	700	1.22	6.7	21,697	26.6	
		B		5	2	*	**	1.12	5.5	20,708	27.0	
		2055A	11-29-65	5	2	10	600	1.30	6.7	19,390	24.7	
		B		5	2	*	**	1.15	5.3	20,170	25.9	
		2056A	11-30-65	5	2	140	1400	0.94	6.6	24,510	32.2	
		2057A	12-1-65	5	2	140	1400	0.95	7.3	24,746	31.5	
		B		5	2	*	**	1.02	6.0	22,618	30.9	

Table 5-8 (Cont.)

Series	Purpose	Test Y-2	Date	Test Config.	Duration (sec)	Hdwe. & Prop. Temp. (°F)	Start Tank Press. (psia)	Time to TC Ignition (sec)	Prop. Preflow (lb)	Turbine Speed at TC Ign. (rpm)	Oxidizer Postflow (lb)	Results
5 (cont.)		2058A	12-2-65	5	2	140	1200	0.97	6.3	23,760	31.9	
		B		5	2	*	**	1.05	5.8	22,700	30.9	
		2059A	12-2-65	5	2	140	1000	1.02	7.1	23,030	31.1	
		B		5	2	*	**	1.00	4.7	22,140	30.5	
		2060A	12-3-65	5	2	140	800	1.05	5.6	22,070	30.7	
		B		5	2	*	**	1.03	5.0	21,972	30.9	
		2061A	12-4-65	5	2	90	1100	1.00	6.5	24,330	30.6	
		B		5	2	*	**	1.05	5.9	22,840	31.2	
		2062A	12-5-65	5	2	90	900	1.06	6.5	21,940	30.6	
		B		5	2	*	**	1.07	6.2	22,230	30.3	
		2063A	12-6-65	5	2	90	800	1.08	5.6	22,311	30.3	
		B		5	2	*	**	1.08	5.8	21,792	30.5	
		2064A	12-8-65	5	2	90	700	1.14	5.9	21,020	28.0	
		B		5	2	*	**	1.10	5.5	22,140	28.3	
		2065A	12-9-65	5	2	32	1000	1.09	4.8	21,980	27.7	
		B		5	2	*	**	1.08	3.2	20,990	27.1	
		2066A	12-9-65	5	2	32	800	1.15	5.0	20,820	27.8	
		B		5	2	*	**	1.08	3.4		25.7	
		2067A	12-10-65	5	2	32	700	1.18	4.4	20,300	26.5	
		B		5	2	*	**	1.12	4.2	20,690	26.6	
		2068A	12-11-65	5	2	32	600	1.25	5.0	19,440	25.9	
		B		5	2	*	**	1.15	3.8	20,030	26.7	
6	Determine effectiveness of using the OFPS to command POSV at extremes in environmental conditions and OFPS settings.	2069A	12-12-65	6	2	10	600	–	41.5***	23,070***	29.4***	414 psig OFPS make. Extremely long oxidizer lead.
		B		6	2	*	**	1.89	33.6	24,698	29.0	
		2070A	12-13-65	7	2	140	1400	1.09	7.1	25,050	31.8	368 psig OFPS make. Note 1 and 2 apply.
		B		7	2	*	**	1.54	22.1	25,772	32.7	
7	To demonstrate safe turbine pump shutdown capability when subjected to various malfunction conditions.	2071	12-14-65	5	2	60	1000					Turbine peaked at 26600 rpm. Unit shuts down at approx. 2.4 secs. due to depletion of fuel in fuel start tank.
		2072	12-14-65	5	1.3	60	1000					Turbine speed up to 75% of design speed in 0.6 secs. Electronic gate cut-off (29500 rpm) at 0.91 secs. Maximum overshoot speed 32000 rpm.

Table 5-8 (Cont.)

Series	Purpose	Test Y-2	Date	Test Config.	Duration (sec)	Hdwe. & Prop. Temp. (°F)	Start Tank Press. (psia)	Time To TC Ignition (sec)	Prop. Preflow (lb)	Turbine Speed at TC Ign. (rpm)	Oxidizer Postflow (lb)	Results
7 (cont.)		2073	12-15-65	8	2	60	1000	0.86	2.4(F)	19,645	28.3	Fuel valve opens sooner – 0.50 sec (in approx. 1/2 the time). 2.4 lb fuel lead in the thrust chamber. OMPS/OFPS circuit closed intentionally on start. Simulates an OMPS or OFPS failure in the switch made position.
		2074	12-15-65	4	1.6 Malf.	60	1000	–	–	–	29.6	Test reproduced the malfunction condition of 2048. All hardware identical including the electronic gate.
		2075	12-16-65	4	1.4 Malf.	60	1000	–	–	–	29.7	Electronic gate tripped due to OFPS chatter. Same as 2074 except an electrical EMI suppression circuit was added to the POSV lock-in relay.
8	Determine the effects of using an ASA J-box on T/P shutdown.	2076A	12-28-65	9	14.6	60	1000	–	–	–	30.5	Intended 60-sec test terminated prematurely because of fire in cell caused by leaking oxid and fuel by-pass valves – no damage. Note 1 applies.
		2077A	12-29-65	10	60	60	1000	–	–	–	30.3	Closing times not significantly altered by the influence of the ASA J-box. Oxidizer valve burst disc used on phase A. No discs on phase B, C and D.
		B			15	*	**				31.1	
		C			2	*	**				30.2	
		D			2	*	**				30.1	
9	Recheck turbine pump operation	See Tests 2078/ 2079 Series 10		10	60	60	1000	–	–	–		Turbine pump operation within prescribed specification limits
10	Demonstrate safe turbine pump shutdown with various malfunction conditions.	2078	12-30-65	10	62.6	60	1000	0.96	0.7(0)	21,600	32.9	Safe shutdown-behavior similar to test 2071. Fuel depletion.
		2079	12-30-65	10	61.8	60	1000	0.96	1.1(0)	21,780	–	Safe shutdown – behavior similar to test 2072. Oxidizer depletion.

Table 5-8 (Cont.)

Series	Purpose	Test Y-2	Date	Test Config.	Duration (sec)	Hdwe. & Prop. Temp. (°F)	Start Tank Press. (psia)	Time to TC Ignition (sec)	Prop. Preflow (lb)	Turbine Speed at TC Ign. (rpm)	Oxidizer Postflow (lb)	Results
11	Turbine pump teardown and inspection. Conversion of cell to low suction pressure configuration.	–	–	–	–	–	–	–	–	–	–	
12.	Low pump suction pressure start transient tests – tests 2080 through 2085 were conducted during the period 18 January through 27 January 1966 to modify and calibrate the low suction pressure propellant feed system.	–	–	–	–	–	–	–	–	–	–	
13	OFPS durability test system calibration – tests 2086 through 2089 were conducted during the period 1 February through 9 February 1966 to adjust the turbine pump steady state speed to provide maximum oxidizer feed pressure fluctuations (21,500 (± 500) rpm).	–	–	–	–	–	–	–	–	–	–	
14	Determine pressure pulse attenuation resulting from an orificed "OFP" line configuration.	2090	2-11-66	10	30	60	1000	–	–	–	–	Speed 21041 rpm. "OFP" peak-to-peak 61.4 ± 8.3 psig at 554 psig nominal. No orifice in OFPS line.
		2091	2-12-66	10	30	60	1000	–	–	–	–	"OFP" peak-to-peak 56.1 (± 4.0) psig at 554 psig nominal. With 0.050" dia. orifice installed in OFPS line.
		2092	2-12-66	10	30	60	1000	–	–	–	–	"OFP" peak-to-peak 47.1 (± 3.2) psig at 554 psig nominal. With 0.030" dia. orifice installed in OFPS line.
15	OFPS durability tests – these tests are discussed in Section 5.7.	–	–	–	–	–	–	–	–	–	–	

Table 5-8 (Cont.)

Series	Purpose	Test Y-2	Date	Test Config.	Duration (sec)	Hdwe. & Prop. Temp. (°F)	Start Tank Press. (psia)	Time to TC Ignition (sec)	Prop. Preflow (lb)	Turbine Speed at TC Ign. (rpm)	Oxidizer Postflow (lb)	Results
16	Restore power level of T/P and check performance	2102A B	2-24-66	10	60 2	60 *	1000 **	–	–	–	–	Speed low. Gas generator mixture ratio too high. Reventuri.
		2103A B	2-25-66	10	60 2	60 *	1000 **	–	–	–	–	Unit satisfied all operational specification requirements.
	Determine effect of varying propellant temperature, start tank loading, and OFPS setting on the turbine pump start transient with no lock-in relay in the POSV command circuit.	2104	2-26-66	4	2	60	1000	1.15	10.9	23,800	30.6	Fuel valve repeatedly shuttled On/Off 3 times. Used an oxid burst disc, 8247-475-481-1 harness and ASA J-box. OFPS actuation setting: 386 psig.
		2105	3-1-66	7	2	140	1400	1.13	12.7	25,670	35.4	OFPS actuation setting 368 psig. Same as test 2104.
		2106	3-2-66	6	2	140	1400	1.35	22.4	27,040	34.6	OFPS actuation setting 414 psig. Same as test 2104.
		2107	3-3-66	6	2	10	600	–	36.5***	20,500***	35.6	Main fuel propellant valve never opened. OFPS actuation setting 414 psig. Same as test 2104.
17	Repeat of test 2055, to establish minimum total impulse capability	2108A B	3-3-66	5	2 2	10 *	600 **	1.42 1.42	4.9 3.5	21,050 20,880	32.3 29.6	Notes 1 and 2 apply. Used 8247-475481-1 harness and ASA J-box.
18	To determine safe turbine pump operation and establish start transient characteristic during various start tank malfunction conditions.	2109	3-4-66	5	0	60	(F)1000 (O) 500	–	–	–	–	Fuel gas generator valve did not open within 7 seconds. Found engine ground inadvertently removed. Nominal loading in fuel start tank. No load in oxid start tank. Used 8247-475481-1 harness and ASA J-box. No burst disc.
		2110	3-5-66	5	0	60	(F)1000 (O) 500	–	–	–	–	Failed to attain a start - conditions similar to test 2109. Suspect fuel GG valve hangup - remove fuel GG valve and install another. Same as test 2109.

Table 5-8 (Cont.)

Series	Purpose	Test Y-2	Date	Test Config.	Duration (sec)	Hdwe. & Prop. Temp. (°F)	Start Tank Press. (psia)	Time to TC Ignition (sec)	Prop. Preflow (lb)	Turbine Speed at TC Ign. (rpm)	Oxidizer Postflow (lb)	Results
18 (cont.)		2111	3-5-66	5	5.8	60	(F)1000 (O) 500	–	–	–	–	Failed to complete start transient due to operator error. Same as test 2109. Installed new fuel GG valve S/N 109.
		2112	3-7-66	5	3.0	60	(F) 500 (O)1000	3.0	–	–	–	Programmed shutdown at 75% of design speed (18000 rpm). Nominal loading in oxidizer start tank. No load in fuel start tank. Used 8247=475481-1 harness and ASA J-box.
		2113	3-8-66	5	11.4	60	(F)500 (O)500	10.9	–	–	–	Conducted a normal suction pressure – Start with cutoff at 75% of design speed. No loading in either the fuel or oxid. start tanks. Used 8247-475481-1 harness and ASA J-box.
		2114	3-9-66	5	13.0	60	(F)1000 (O) 500	12.2	–	–	–	Repeat test 2111. Shutdown initiated at 75% of design speed. Same as test 2109.

*Hardware (only) permitted to normalize to ambient temperature for Phase B start.

**Phase B start tank loading equal to the Phase A regenerated condition.

***OFPS did not actuate, values taken at shutdown signal.

(F) Fuel

(O) Oxidizer

Table 5-8 (Cont.)

Note 1. **Phase A starts with oxidizer burst** disc located downstream of oxidizer propellant valve.
Phase B starts without burst disc.

Note 2. Propellant lines are free of liquid prior to start (both phases).

(a) From gas generator valves downstream to gas generator injector.

(b) From oxidizer propellant valve downstream to thrust chamber injector.

Configurations:

1. XLR 81-BA-11 main fuel and oxidizer valves. Fuel valve actuated by a simulated OMPS in an orificed oxidizer discharge pressure line (setting 430 psig).
2. YLR 81-BA-11 main fuel and oxidizer valves. Simulated engine configuration (thrust chamber). Fuel valve actuated by OMPS (setting 32 psig). Monitored OFPS located on oxidizer valve (setting 235 psig). POSV pip monitored.
3. XLR 81-BA-13 main fuel and oxidizer valves. Other conditions same as configuration 2.
4. YLR 81-BA-11 main fuel and oxidizer valves. Simulated engine configuration (thrust chamber). Fuel valve actuated by OFPS located on oxidizer valve (setting 386 psig). Monitored OMPS (32 psig) and POSV pip.
5. YLR 81-BA-11 main fuel and oxidizer valves. Simulated engine configuration (thrust chamber). Fuel valve actuated by OMPS (setting 32 psig) and OFPS (setting 386 psig) connected in parallel. OMPS/OFPS pip monitored independently. POSV pip monitored.
6. Same as configuration 4 except OFPS setting is set to make at 414 psig.
7. Same as configuration 4 except OFPS setting is set to make at 368 psig.
8. YLR 81-BA-11 main fuel and oxidizer valves. Simulated engine configuration (thrust chamber). Fuel valve actuated on start signal i.e. OMPS/OFPS made on start.
9. Same as configuration 1 except OMPS setting is 710 psig. Used 8247-475481-1 harness, 8247-472483-1 relay box and ASA J-box.
10. Same as configuration 1 except OMPS setting is 550 psig. Used 8247-475481-1 harness, 8247-472483-1 relay box and ASA J-box.
11. Same as configuration 1 except OMPS setting 640 psig. Used 8247-475481-1 harness, 8247-472483-1 relay box and ASA J-box.
12. YLR 81-BA-11 main fuel and oxidizer valves. Circulating propellant system using orifice simulated engine impedance. Fuel valve actuated electrically on start. No burst discs on fuel or oxidizer system.

Table 5-9

OFF LOADED START TANK TEST RESULTS

Conditions	Test Number	Prop. and Hardware Temp. (° F)	Initial Start Tank Gas Press. (psia)	Time to 75% Turbine Speed (sec)	Time to Ignition (sec)	Turbine Speed at Ignition (rpm)
Fuel Start Tank Empty	2112	60	1000 (ox) 500 (F)	2.948	2.956	19,500
Both Start Tanks Empty	2113	60	500 (ox) 500 (F)	11.235	10.806	18,300
Oxidizer Start Tank Empty	2114	60	500 (ox) 1000 (F)	12.780	12.191	18,100

Table 5-10

START TRANSIENT VARIATIONS DUE TO CONFIGURATION CHANGES

OMPS Commanding							
Configuration	Propellant & Hardware Temperature (°F)	Initial Start Tank Pressure (psia)	Time to Thrust Chamber Ignition (sec)	Oxidizer Preflow (lb)	Turbine Speed at Thrust Chamber Ignition (rpm)	Time to OMPS Actuation (sec)	Comments
Relay Box	60	1000	1.083	6.8	22,270	0.972	
Junction Box	60	1000	1.040	7.5	23,400	0.930	Data calculated from an OFPS command run.
OFPS Commanding							
Configuration	Propellant & Hardware Temperature (°F)	Initial Start Tank Pressure (psia)	Time to Thrust Chamber Ignition (sec)	Oxidizer Preflow (lb)	Turbine Speed at Thrust Chamber Ignition (rpm)	Time to OFPS Actuation (sec)	Oxidizer Feed Pressure at OFPS Actuation (psig)
Relay Box OFPS Actuation = 391 psig	60	1000	1.275	15.2	24,050	0.927	407
Relay Box OFPS Actuation = 414 psig	10	600	> 2	41.5 (at 2 sec.)	23,070	Did Not Lock In	–
Relay Box OFPS Actuation = 368 psig	140	1400	0.985	7.1	25,050	0.918	370
J-Box OFPS Actuation = 391 psig	60	1000	1.154	10.9	23,800	1.043	380
J-Box OFPS Actuation = 414 psig	10	600	> 2	36.5 (at 2 sec.)	20,500	Did Not Lock In	–
J-Box OFPS Actuation = 368 psig	140	1400	1.129	12.7	25,670	1.018	378

Table 5-11

EFFECT OF ASA J-BOX ON MODIFIED ENGINE SHUTDOWN TRANSIENT

	With ASA J-Box[1]		Without ASA J-Box[2]	
	Average	Std. Dev.	Average	Std. Dev.
Time to Valve Close (sec)				
Oxidizer GG	0.109	±0.0047	0.109	±0.0046
Fuel GG	0.157	±0.0049	0.162	±0.0061
Main Fuel Valve	0.031	±0.0039	0.030	±0.0033

NOTES: (1) Turbine pump runs 2076 and 2077 – 5 tests.
(2) Turbine pump runs 2040 through 2068 – 36 tests.

5.5 START SYSTEM TEST PROGRAM

The gas generator start system tests were conducted to determine the start system ignition and combustion performance characteristics at altitude over a range of possible start conditions. Also to be evaluated were the effects of storage for 28 days at sea level plus 5 days at altitude during which time propellants were maintained in the start system, simulating a maximum mission period (pad hold time plus mission orbit life). Results to be obtained were the effect of the primary influencing and independent parameters of feed pressure, temperature, and resultant mixture ratios on gas generator altitude ignition and combustion performance, and the effect of combined long-term sea level and altitude storage on ignition. The start system tests were conducted at BAC (Test Cell X-2), from 20 November 1965 through 12 March 1966.

5.5.1 Test Configuration

Test components used in the gas generator ignition tests consisted of fuel and oxidizer start tanks, fill valves, venturi assemblies, solenoid valves, and a gas generator turbine manifold assembly.

The above components were geometrically located as in the complete engine installation using production hardware. That part of the engine used for the start system tests is shown schematically by the solid lines in Fig. 5-10. The start system used a "blow down" operation during the test series allowing a simulation of normal engine operation for a period of approximately 0.7 to 0.8 seconds duration at which time turbine pump recharge of the start tanks is normally initiated.

An altitude duct was welded to the turbine manifold downstream of the turbine nozzles to maintain altitude conditions during propellant lead and ignition by providing an additional 5.5 cubic feet of evacuated volume. A trap door with a seal was provided on the end of the duct to evacuate the system prior to the start of a test. The trap door would blow open when the duct pressure exceeded atmospheric pressure. The duct also conducted the fuel rich exhaust gas away from the test cell. Cryogenic

cold traps were installed in the exhaust system to minimize evaporated propellant contamination of the equipment. The gas generator test assembly was completely enclosed in an environmental chamber for hardware and propellant temperature conditioning as indicated in Table 5-12.

Instrumentation used to meet the program objectives is shown in Fig. 5-11. High-response pressure instrumentation was used to monitor propellant injection vapor pressures, propellant vapor reactions, and ignition peak pressures and pressure spikes. This instrumentation was utilized not only for generator chamber pressure during all tests but also for fuel and oxidizer injector manifold feed line pressures during runs 2420 through 2428. Figures 5-12 and 5-13 are photographs of the test cell, showing hardware installation and component configuration.

During the test series, gas generator solenoid valve S/N 109 was replaced by valve S/N 112. This replacement was due to failure of valve 109 to open (refer to Section 8 for valve failure discussion).

5.5.2 Test Results

Safe and reliable ignition occurred on all tests except 2397 when fuel solenoid valve S/N 109 failed to open. No ignition delay or spiking deleterious to the engine start transient or the component integrity was noted during any of the tests, including the effects from the malfunction tests. The significant start transient parameters and values thereof for each test are presented in Table 5-12. Graphical presentation of significant data and results are presented in Figs. 5-14 through 5-19.

Figure 5-14 shows peak and steady state gas generator chamber pressure as a function of initial start tank pressure. The steady state chamber pressure for these tests is defined as that pressure occurring at a standard time of 0.5 seconds after start signal. This time interval insures dissipation of the energy from accumulated propellant preflow.

The significant results shown in Fig. 5-14 are as follows:

a. The peak and steady state generator chamber pressures are not significantly affected by either altitude or propellant and hardware temperatures.

b. The peak generator chamber pressure increases with increasing start tank pressure. The relation shown indicates that a factor other than the propellant flow rate increase associated with the higher tank pressures is involved. However, the peak generator pressure at maximum flight operating start tank pressures are well below the structural capability of the gas generator.

Figure 5-15 presents a summary of the effects of temperature on the time to ignition, and generator chamber peak pressures after start signal. The time for propellant manifold fill and subsequent liquid injection for both propellants are shown for baseline purposes. Although the data presented is for a nominal start tank pressure of 1,000 psia, analysis of data at other tank pressures shows a good correlation at both sea level and altitude between time to generator chamber peak pressure and propellant flow rate (which is a direct function of the start tank pressures). The time to ignition at altitude conditions correlates reasonably well with propellant flow rate; however, at sea level conditions, the time to ignition was approximately constant for a specific temperature within the start tank pressure range of 600 to 1,400 psia. Present conclusions indicated from Fig. 5-15 and related data are:

a. Electrical power is applied simultaneously to the fuel and oxidizer generator solenoid valves to give a normal engine start sequence that results in a fuel lead of approximately 80 to 150 milliseconds (due to the higher fuel flow rate) over the maximum range of start tank operating pressures.

b. The tests at maximum temperature show ignition at approximately the time of liquid fuel injection, indicating that fuel reacts with the oxidizer prior to complete filling of the oxidizer manifold. The data indicates a trend to suppress this fuel-oxidizer reaction as temperature is decreased such that at colder temperatures, ignition occurred nearly coincident with the calculated time of oxidizer liquid injection.

c. The time from liquid propellant injection to generator chamber peak pressure correlates reasonable well with fuel injection time at high temperatures and with oxidizer injection time at low temperatures over the test range of start tank pressures.

d. An overall reduction of about 30 percent in time to peak gas generator chamber pressure was noted when comparing hot-test to cold-test data at the same start tank pressure.

e. Hot tests resulted in similar times from start signal to generator chamber peak pressure at altitude and sea level. However, altitude peak pressure was achieved prior to sea level peak pressure during the rated- and cold-temperature tests.

Figure 5-16 compares typical high-temperature ignition characteristics at sea level and altitude. The low pressure rise prior to liquid injection of either propellant was noted consistently during sea level high-temperature tests, and randomly during rated- and cold-temperature tests at sea level. The temperature obtained from a gas thermocouple inserted into the combustion chamber at the injector during the sea level tests also shows a propellant reaction correlation with the chamber pressure rise. This initial reaction at sea level pressure, slightly above the propellant vapor pressures, is attributed to vapor or condensed vapor reaction within the generator chamber (calculations indicate that liquid injection had not yet occurred by the time of this initial reaction). The subsequent decay in pressure and temperature is attributed to a combination of suppressing the vapor flow due to the chamber pressure rise and quenching of the reaction when the liquid enters the combustion chamber.

Propellant injection tests were conducted to determine the gas generator chamber pressure, temperature, and propellant reaction transient characteristics at altitude conditions. The results of a fuel injection test, an oxidizer injection test, and a normal fuel lead ignition test are presented in Fig. 5-17, which compares the low range generator chamber pressure and chamber combustion gas temperatures for the three tests. A reaction occurred during the normal start test as evidenced

by the bi-propellant initial pressure rise above the single propellant vapor pressure, and the temperature increase prior to liquid oxidizer injection into the chamber. A fuel injection quenching effect was noted in temperature but less severe than observed at sea level and generator chamber pressure did not decay. Ignition for this test is defined as 0.074 seconds after start signal, coincident with the generator chamber pressure rise. This vapor ignition phenomena was observed during only two +140°F tests at altitude.

The effects of varying propellant lead times at altitude on the peak gas generator chamber pressure is shown in Fig. 5-18. Two temperature conditions were tested, but the emphasis was on the +10°F condition to determine effects of increased propellant accumulation and probable longer hypergolic propellant reaction time before ignition. A maximum peak generator chamber pressure was observed at the nominal fuel lead condition; increasing the fuel lead decreased the peak pressure. Reducing the fuel lead correspondingly reduced the preignition accumulation of fuel, reduced the generator peak pressure, and at simultaneous propellant injection the pressure overshoot essentially was eliminated. Increasing oxidizer lead with the fuel-rich operation of the gas generator resulted in increasing peak pressures to approximately the same levels as observed for the normal fuel lead condition.

Six tests, 2420 through 2425, were conducted with special feed pressure instrumentation to detect possible ignition in the injector manifolds. These tests were conducted at temperatures of +10°F and +140°F, start tank pressures of 800 psia and 1400 psia, and fuel leads of 0.86 seconds to oxidizer leads of 0.55 seconds to induce entry of a propellant in the opposite propellant injector and feed line manifolds. However, there was no indication of combustion or ignition spikes in the injector or feed lines, based on this instrumentation.

The accelerometer data summary is presented in Fig. 5-19 as a function of initial start tank pressure. A general increase in "g" loading with increasing temperature and pressure was noted for both the horizontal and axial measurements. Maximum accelerations were observed at the time of generator peak chamber pressures;

however, no significant accelerations were observed which would indicate an adverse ignition shock. The sea level peak accelerations were generally lower than those at altitude, for the same test conditions. Observed vibration frequencies were generally random within the range of 1000 to 18,000 cps and no correlation with either pressure or temperature was obtained.

Safe, reliable ignition occurred on the three ignition tests conducted as part of the storage program. No ignition delay or pressure spikes deleterious to the engine start transient or the component integrity was recorded.

A comparison of the post storage test data with previous test data indicated the following:

a. The peak gas generator chamber pressures were 60 to 80 psi above the normal established during the previous tests at comparable test conditions; however, this difference is considered insignificant with respect to ignition, reliability, and structural integrity.

b. Post storage and post coast test valve opening times increased as shown in Table 5-12 as compared to the base line (test 2426) but were within specification maximums.

c. Calibration flow test of both venturi assemblies indicated flow instability of approximately 1.0 percent compared with acceptance test instability limits of ±0.5 percent. However, prestorage instability base line tests were not performed and consequently it is not possible to assess the storage effects on venturi stability. The gas generator performance evaluation showed no indication of performance or propellant flow instability during the post storage hot fire test for the test duration of 1.0 second. Other factors also effect gas generator performance and hence overall engine thrust level. If the 1.0 percent instability did occur in flight, it would not significantly affect mission performance and sequencing, based on the manner in which all factors are accounted for in present primary propulsion maneuver planning.

5.5.3 Conclusions

Safe and reliable ignition was demonstrated during all altitude tests, and no ignition delay, pressure spikes, or adverse peak pressures were observed which would be detrimental to engine operation or integrity during flight environment. The generator chamber peak pressures are a function of initial start tank pressure and fuel lead time, and appear to be only secondary functions of temperature and ambient pressure. The turbine manifold pressure followed the gas generator chamber pressure characteristics closely at expected slightly lower pressures. The time from start signal to ignition is a function of temperature and initial start tank pressure. Ambient pressure effects on the time to ignition were observed only at low temperatures. A propellant vapor ignition phenomena was observed at sea level; however, this phenomena occurred only at high temperatures during the altitude tests. Generator ignition after 28 days and 5 additional days of storage was reliable and key parameters compared favorable with the prior calibration test and ignition test data.

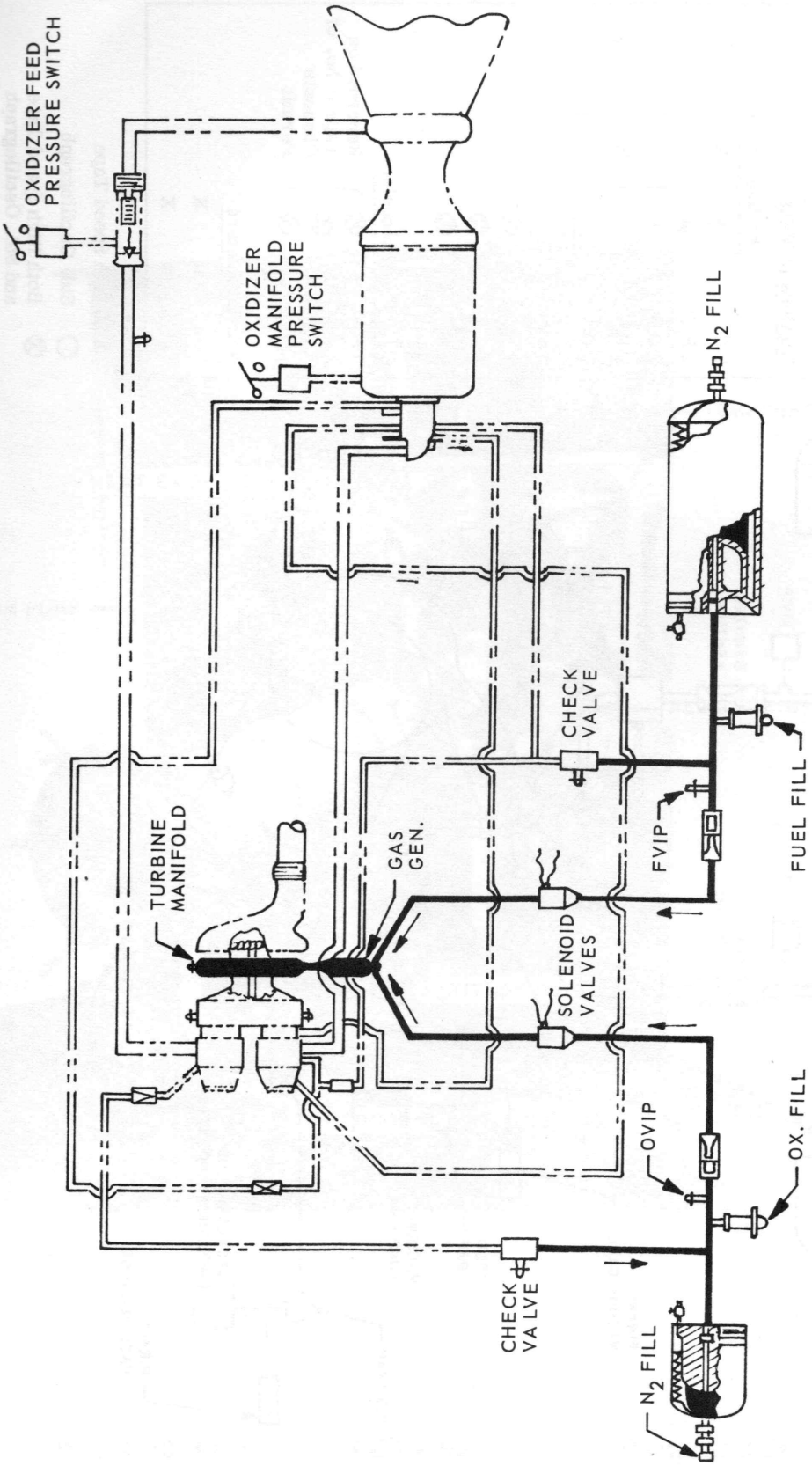


Fig. 5-10 Engine Start System Propellant Flow Diagram

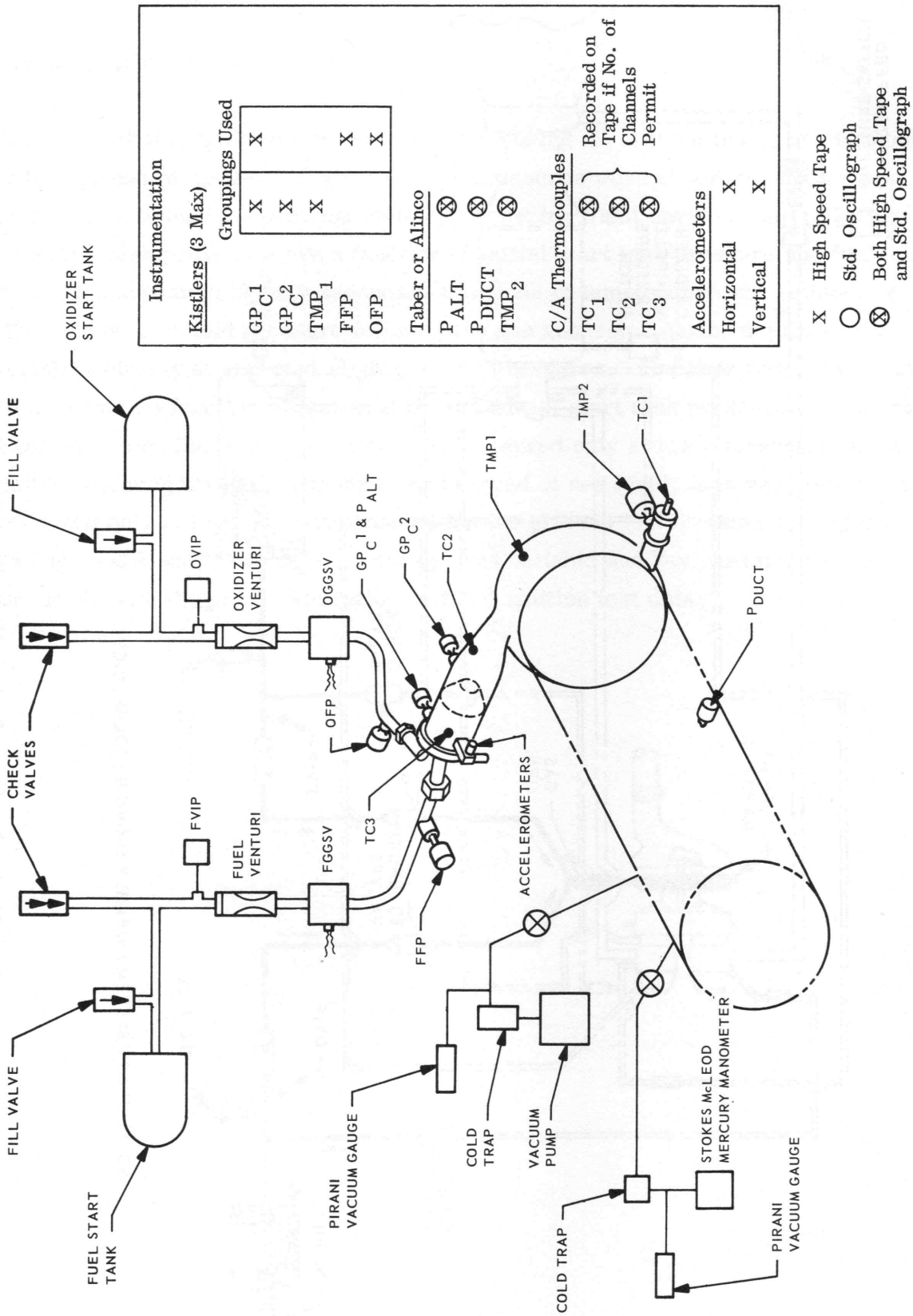


Fig. 5-11 Start System Instrumentation Locations

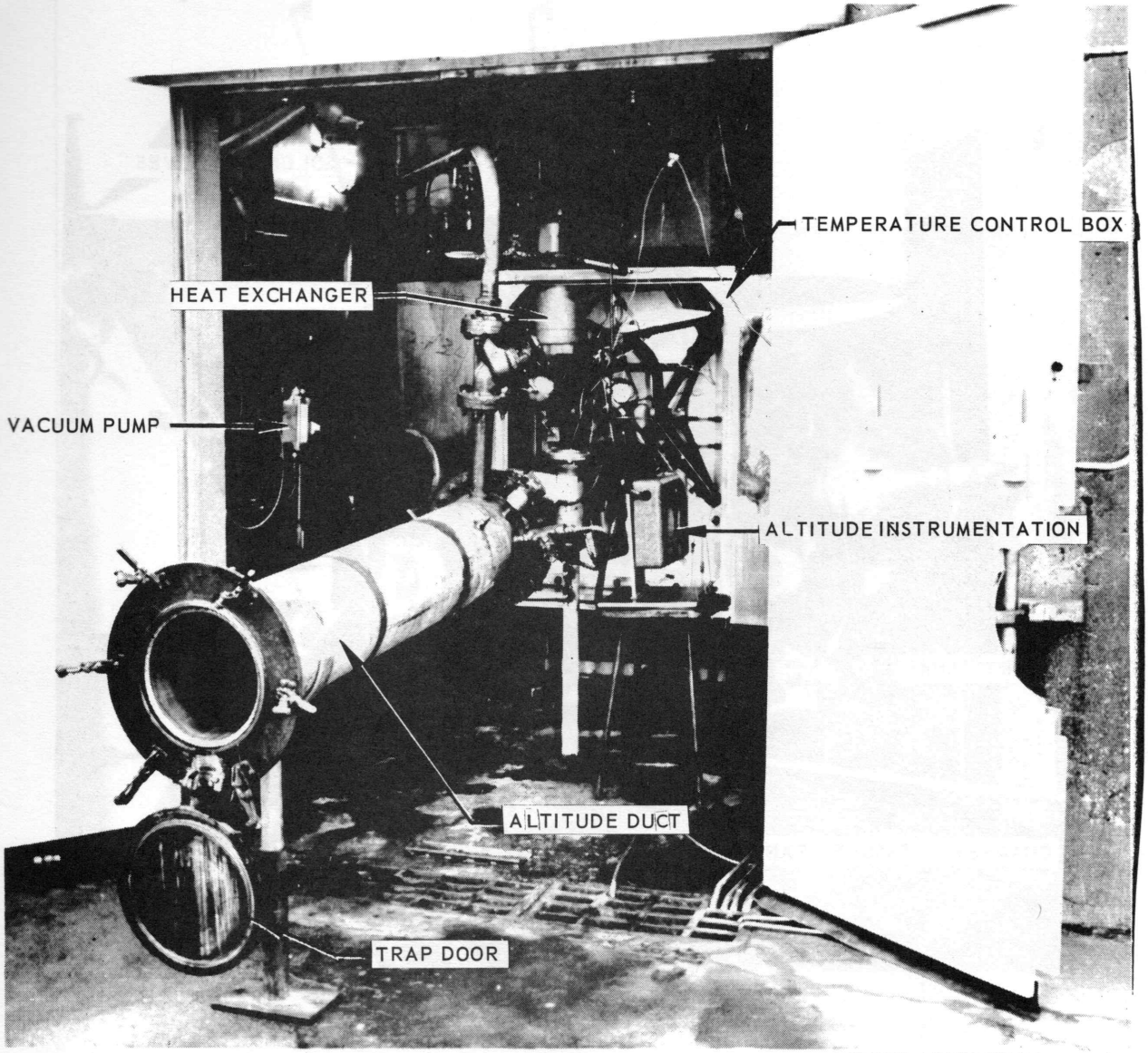


Fig. 5-12 Engine Test Cell – Hardware and Component Installation

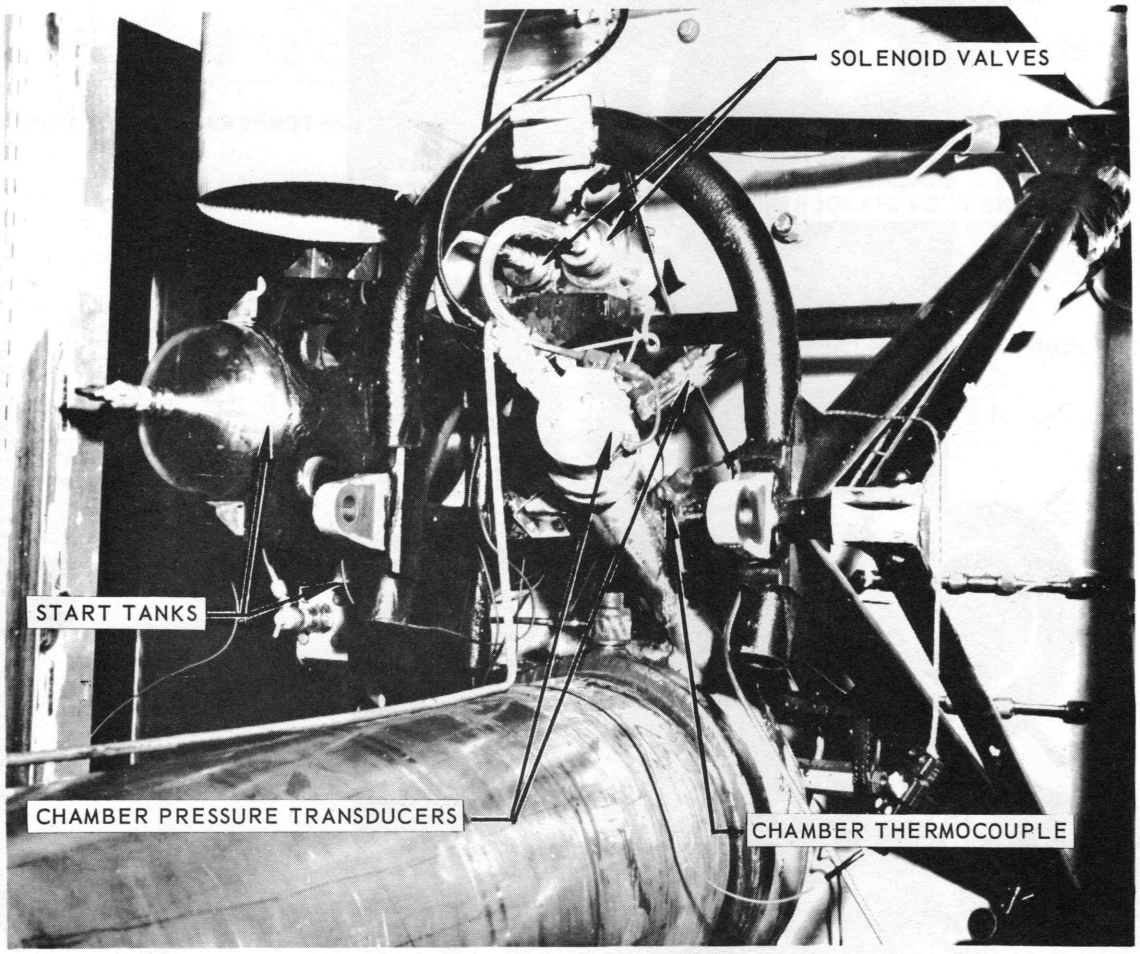


Fig. 5-13 Engine Test Cell – Hardware and Component Installation

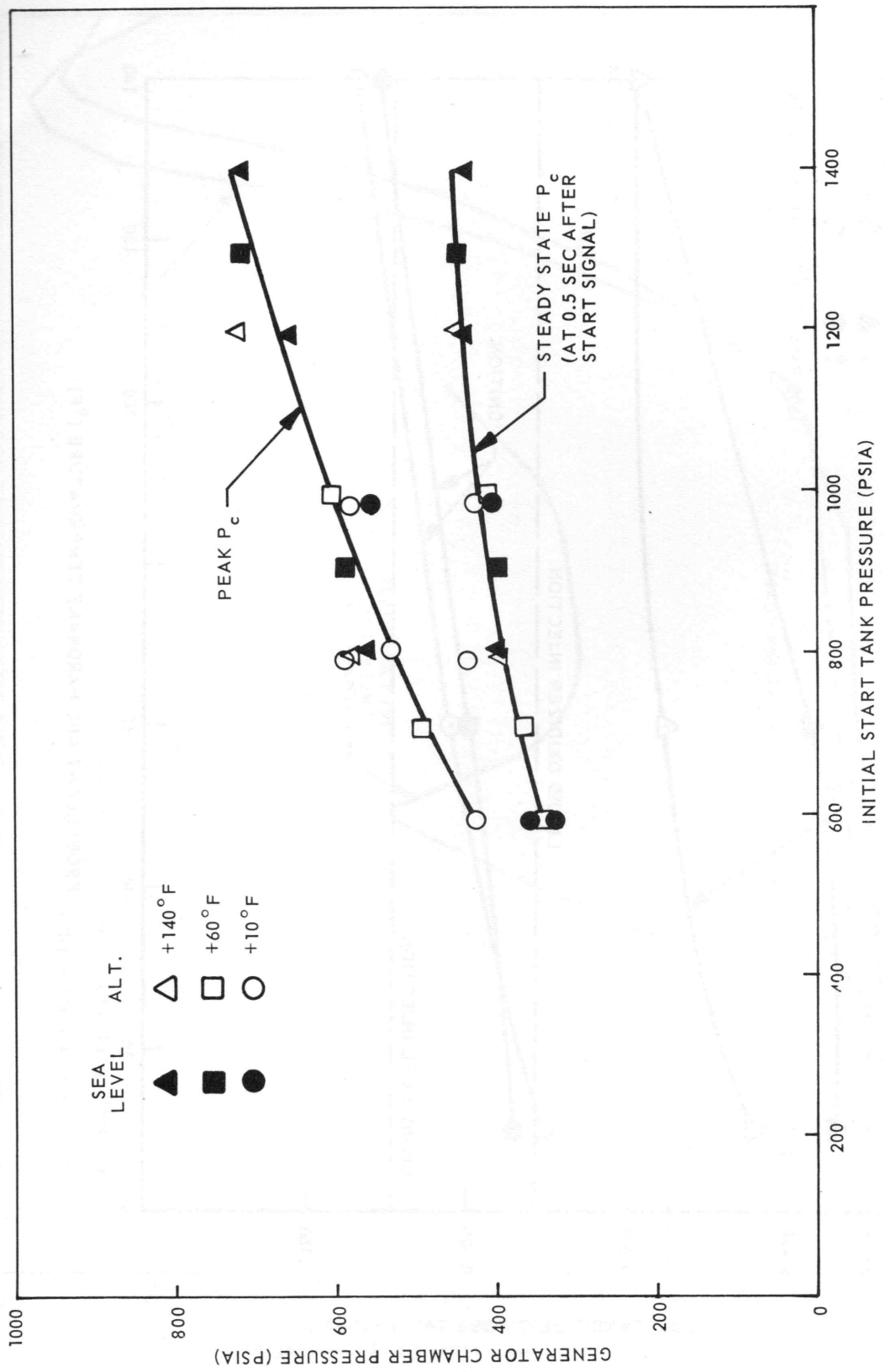


Fig. 5-14 Gas Generator Pressure vs. Start Tank Pressure

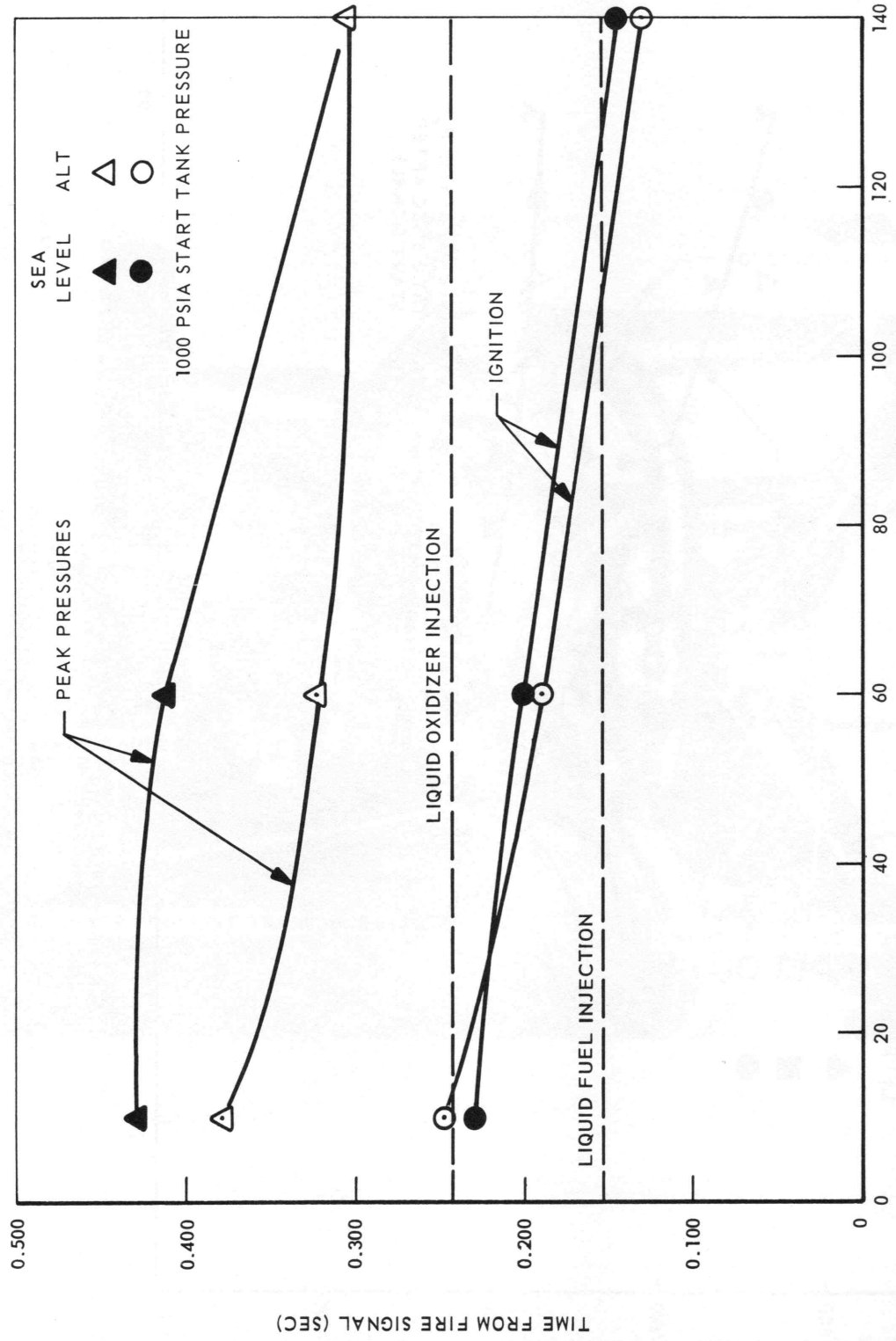


Fig. 5-15 Time to Ignition and Peak Pressures

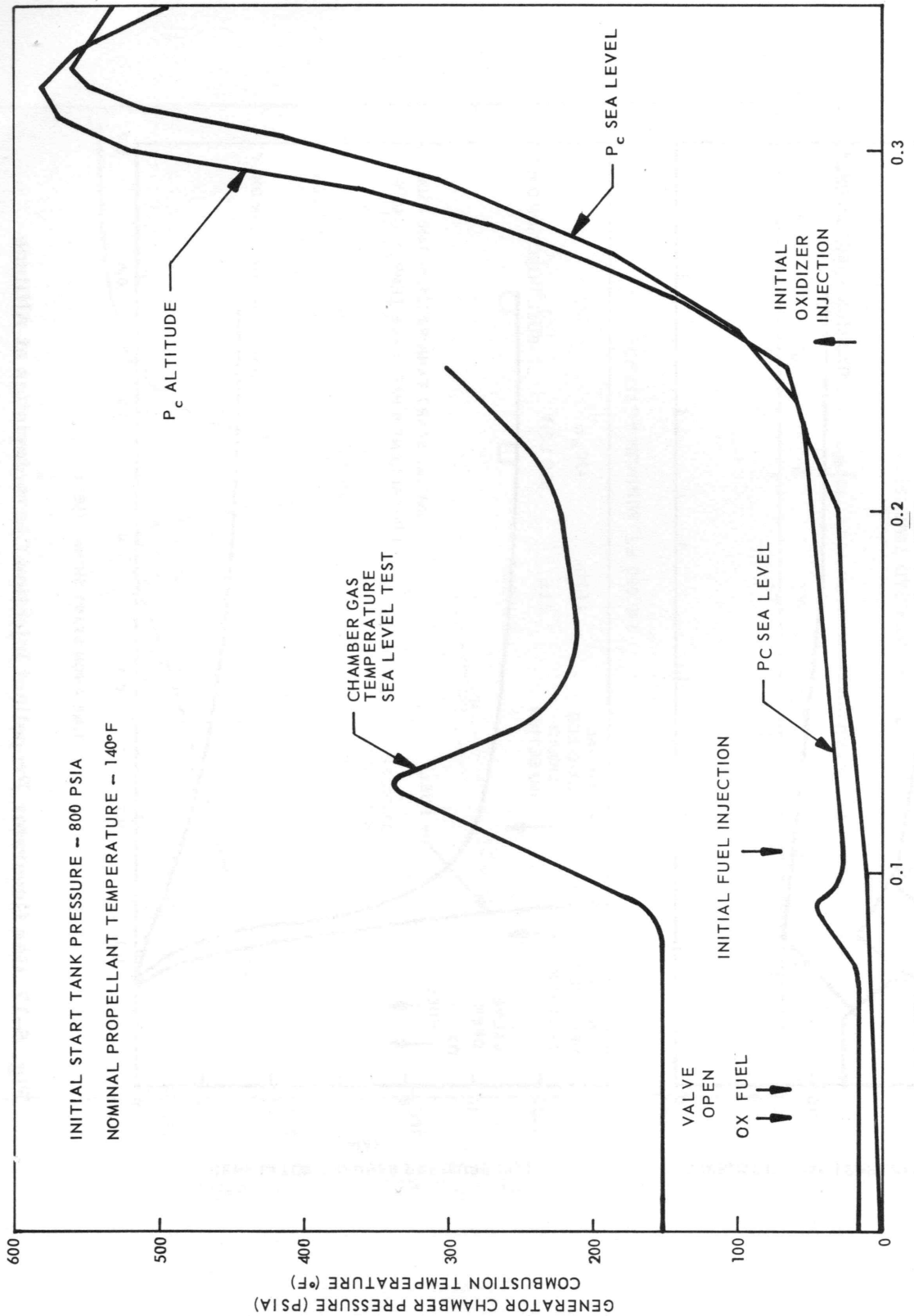


Fig. 5-16 Gas Generator Temperature and Pressure Time Histories

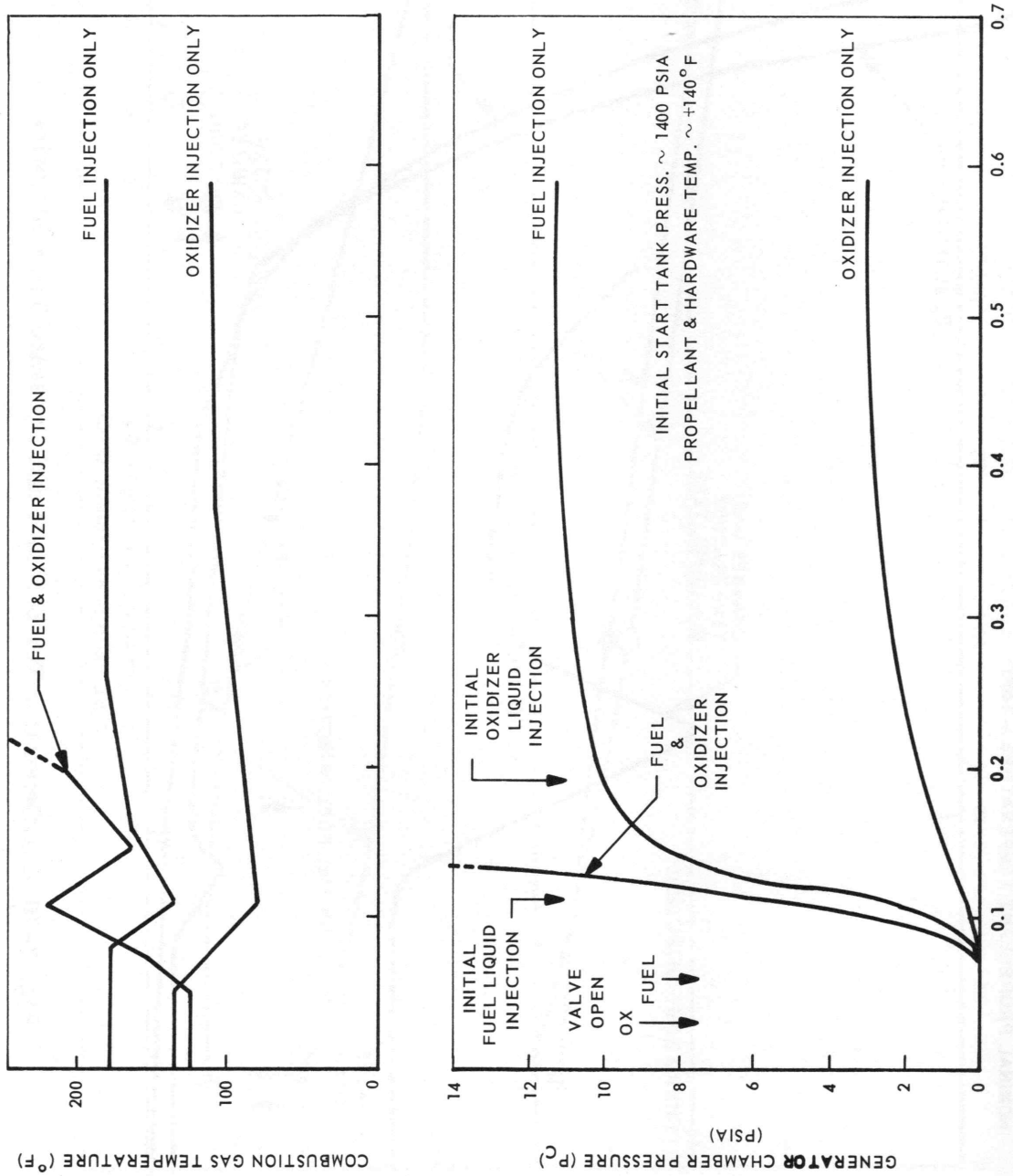


Fig. 5-17 Gas Generator Propellant Injection Characteristics at Altitude

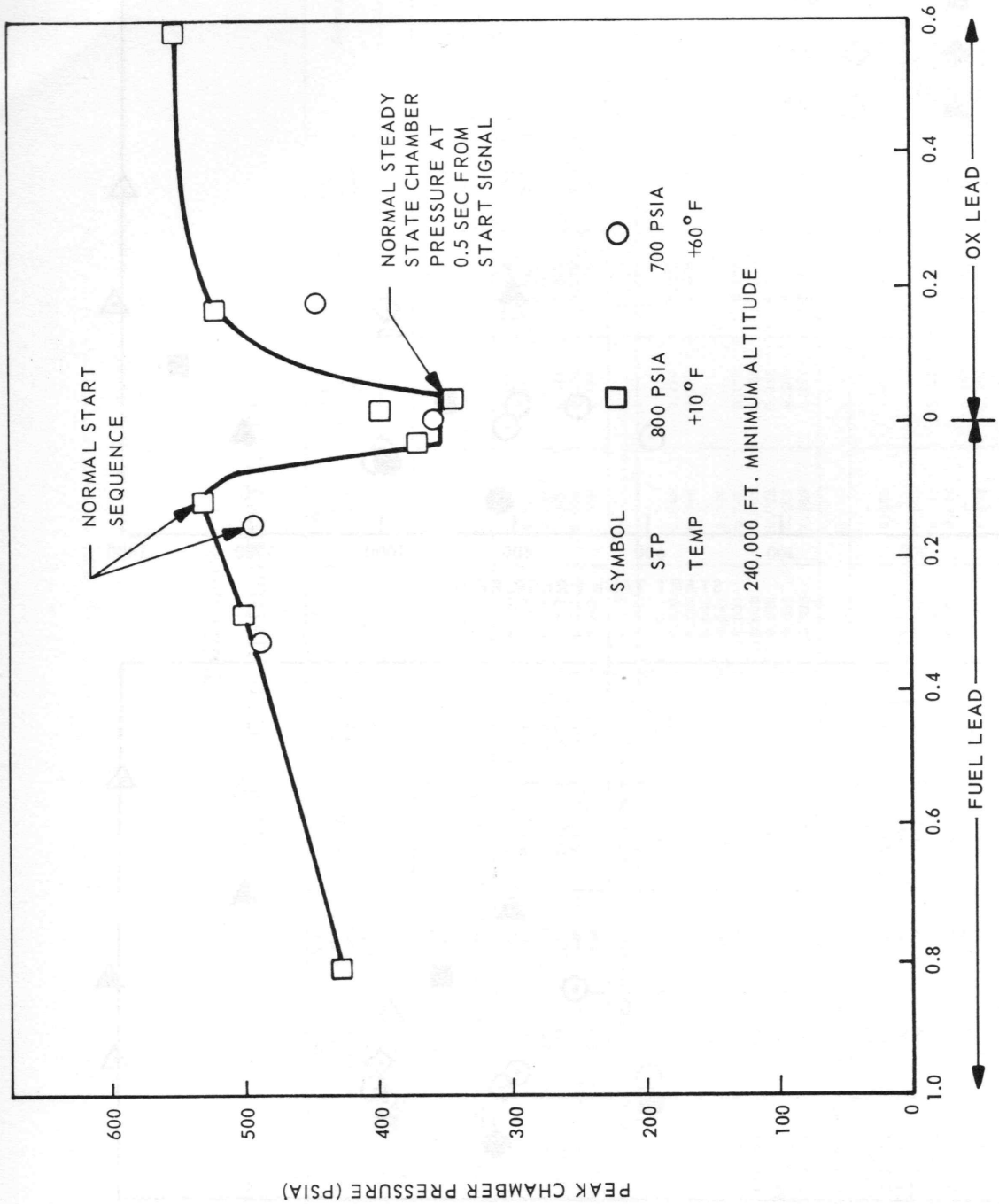


Fig. 5-18 Peak Gas Generator Chamber Pressure as a Function of Propellant Lead Time

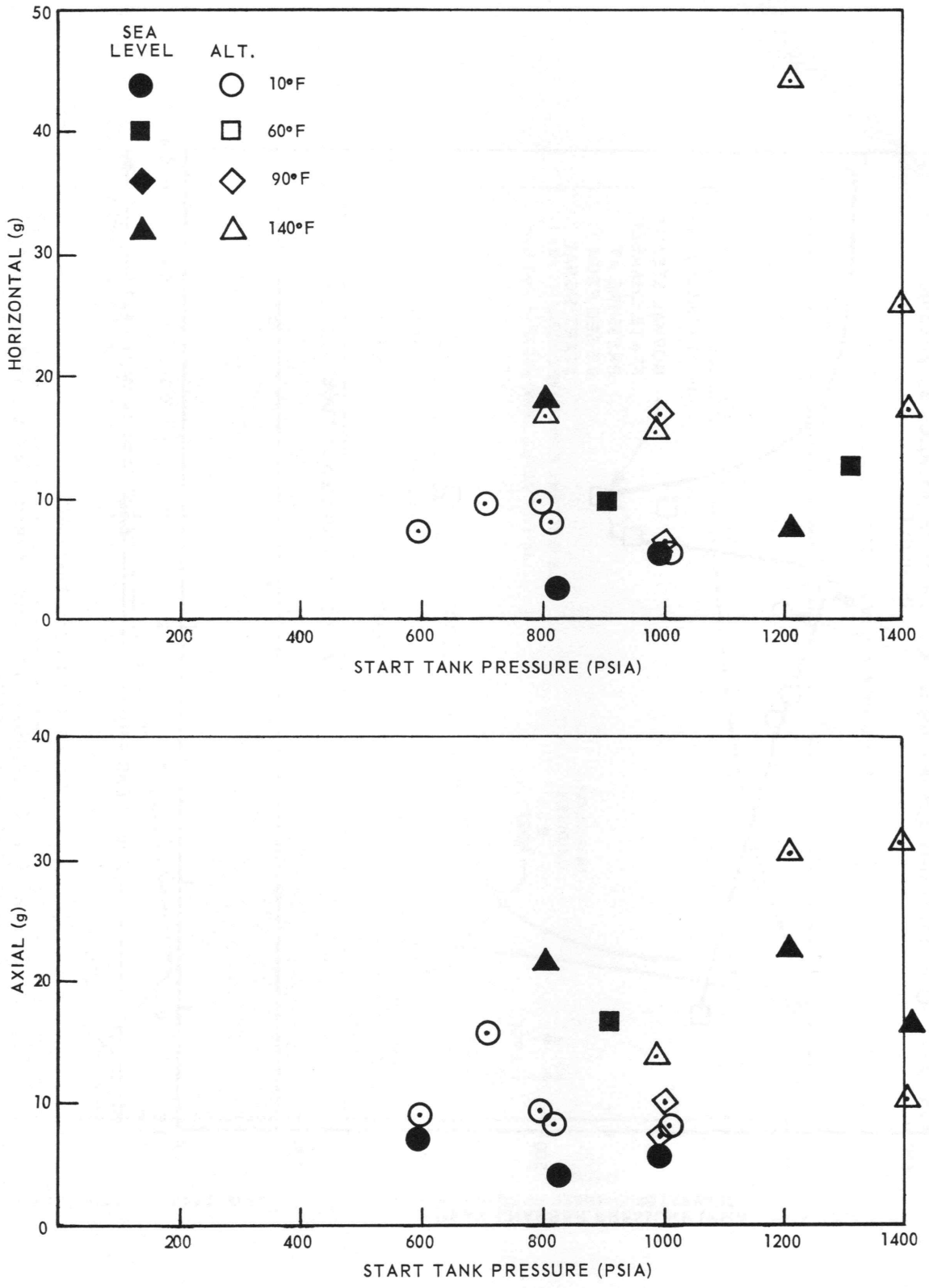


Fig. 5-19 Accelerations at Peak Chamber Pressure

Table 5-12

TURBINE MANIFOLD TEST DATA

X-2 Run No. and Date		Start Tank Press. (psia)	Prop. & Hwe. Temp. (° F)	Altitude	Duration (sec)	Time to GGOV Crack (sec)	Time to GGFV Crack (sec)	Time to GPc Rise (sec)	Time to GPc Peak (sec)	Peak GPc (psia)	Remarks
INSTALLATION CHECK OUT AT SEA LEVEL											
2390	11/20/65	900	60	S.L.	1.10	–	0.036	0.200	0.426	585	
2391	11/21/65	1300	60	S.L.	1.11	0.037	0.039	0.208	0.368	714	
VARIOUS START TANK LOADS AND PROPELLANT TEMP. AT SEA LEVEL											
2392	11/21/65	1000	10	S.L.	1.10	0.028	0.079	0.221	0.476	356	
2393	11/22/65	800	10	S.L.	1.11	0.038	0.050	0.232	0.430	553	
2406	12/08/65	600	10	S.L.	0.6	0.048	0.050	–	–	–	
2407	12/09/65	1400	140	S.L.	0.59	0.032	0.042	0.124	0.319	559	
2408	12/10/65	1200	140	S.L.	0.59	0.041	0.064	0.155	0.270	713	
2409	12/12/65	800	140	S.L.	0.59	0.038	0.055	0.162	0.283	–	
VARIOUS START TANK LOADS AND PROPELLANT TEMP. AT ALTITUDE											
2394	11/26/65	1300	60	240K	1.11	–	–	–	–	–	
2395	11/27/65	1000	60	247K	1.11	0.034	0.049	0.190	0.323	602	
2396	11/28/65	700	60	248.5K	1.11	0.031	0.048	0.230	0.380	491	
2397	12/01/65	1400	140	257K	1.11	0.040	–	–	–	–	GGFV failed to open
2398	12/02/65	1400	140	257K	1.10	0.041	0.049	–	–	–	
2399	12/02/65	1200	140	255K	0.59	0.038	0.048	0.120	0.290	719	
2400	12/03/65	800	140	257K	0.60	0.033	0.047	0.138	0.318	580	
2402	12/05/65	1000	10	255K	0.59	0.034	0.072	0.248	0.378	579	
2403	12/06/65	800	10	254K	0.59	0.032	0.072	0.250	0.368	529	
2405	12/08/65	600	10	254K	0.59	0.029	0.069	0.273	0.453	426	
SIMULATED MALFUNCTION TEST AT ALTITUDE											
2401	12/04/65	800	140	259K	0.59	–	–	–	–	–	Fuel flow only
2410	12/14/65	1400	140	307K	0.6	–	0.058	–	–	–	Fuel flow only (repeat of 2401)
2411	12/15/65	1400	140	284K	0.6	0.041	–	–	–	–	Oxidizer only (repeat of 2397)
2412	12/16/65	1400	140	271K	0.6	0.040	0.053	–	–	–	Repeat of 2398
2413	12/17/65	700	60	272K	0.6	0.028	0.048	0.256	0.380	–	Simultaneous propellant entry
2414	12/17/65	700	60	265K	0.8	0.029	0.049	–	–	–	Fuel lead of 0.32 sec
2415	12/20/65	700	60	280K	0.9	–	–	0.447	0.501	438	Ox. lead of 0.16 sec
2416	12/21/65	700	60	265K	0.8	–	0.037	0.396	0.526	–	Repeat of 2414

Table 5-12 (Cont.)

X-2 Run No. and Date		Start Tank Press. (psia)	Prop. & Hwe. Temp. (° F)	Altitude	Duration (sec)	Time to GGOV Crack (sec)	Time to GGFV Crack (sec)	Time to GPc Rise (sec)	Time to GPc Peak (sec)	Peak GPc (psia)	Remarks
SIMULATED MALFUNCTION TEST AT ALTITUDE (Cont.)											
2417	12/22/65	800	10	272K	0.8	–	0.060	0.118	0.195	397	Simultaneous propellant entry
2418	12/23/65	800	10	277K	0.8	–	0.064	0.394	0.576	497	Fuel lead of 0.32 sec
2419	12/30/65	800	10	310K	0.8	0.030	–	0.424	0.470	518	Ox. lead of 0.16 sec
2420	1/04/66	800	10	323K	0.8	0.033	–	0.248	0.412	588	Normal lead
2421	1/05/66	1400	140	301K	0.8	0.040	–	0.074	0.183	–	Normal lead
2422	1/08/66	800	10	292K	0.8	0.030	0.055	0.253	0.315	1056	Fuel lead of 0.06 sec
2423	1/10/66	800	10	311K	0.9	0.024	0.044	0.195	0.295	490	Ox. lead of 0.06 sec
2424	1/10/66	800	10	317K	1.1	0.028	0.058	0.819	0.868	547	Ox. lead of 0.55 sec
2425	1/11/66	800	10	352K	1.1	0.031	0.055	0.862	1.104	429	Fuel lead of 0.86 sec
28 DAY STORAGE AND 5 DAY COAST TEST AT ALTITUDE											
2426	2/03/66	1000	90	295K	1.0	0.033	0.052	0.100	0.330	677	Pre 28 day storage calibration run
2427	3/06/66	1000	90	307K	1.0	0.041	0.072	0.131	0.342	776	*Post storage test
2428	3/11/66	1000	90	337K	1.0	0.036	0.073	0.116	0.328	738	Post coast test

*5 Day Coast was at an altitude of 80,000 ft (min.)

5.6 SPECIAL ENGINE INSTRUMENTATION TEST PROGRAM

The primary objective of the special instrumentation program was to determine if the modified (oxidizer lead) XLR 81-BA-13 engine start transient dynamic characteristics are similar to or significantly different than the standard YLR 81-BA-11 engine configuration. To accomplish the objective, dynamic instrumentation was recorded during acceptance testing of four YLR 81-BA-11 and three XLR 81-BA-13 engines. Test durations for the YLR 81-BA-11 engines were two 60-second firings; for the XLR 81-BA-13 engines, three 2-second and two 60-second firings. The tests were conducted between 26 December 1965 and 18 March 1966 at Bell Aerosystems Company (Test Cell H-2).

5.6.1 Test Configuration

The test articles were production engines acceptance tested in the horizontal attitude at sea level environment. The engine nozzle extension was replaced with a mass simulator used for engine thrust chamber gimballing and for thrust chamber operation at sea level atmospheric pressure. Engine configurations were in accordance with the normal YLR 81-BA-11 and XLR 81-BA-13 acceptance test requirements as called out in the respective Model Specifications.

Acceleration, force, and pressure measurements were recorded at the following locations which are shown graphically in Fig. 5-20:

a. Accelerometers

- Thrust Chamber Thrust Lug – Longitudinal (A-1)
- Thrust Chamber Thrust Lug – Yaw (A-2)
- Thrust Chamber Thrust Lug – Pitch (A-3)
- Gimbal Ring Assembly – Longitudinal, (A-4) Top
- Gimbal Ring Assembly – Longitudinal, (A-5) Bottom
- Thrust Mount Structure – Longitudinal, Left Gimbal (A-6)
- Thrust Mount Structure – Longitudinal, Right Gimbal (A-7)

Thrust Mount Structure – Longitudinal, Actuator Top (A-8)

Thrust Mount Structure – Longitudinal, Actuator Side (A-9)

Gas Generator – Parallel to Geometric Axis of Generator near Injector Head (A-10)

Engine Test Rig – Longitudinal (A-11)

Engine Test Rig – Pitch (A-12)

Electronic Gate Box – Longitudinal, in rigid area near engine attachment point (A-13)

Pressure Switch Relay Box – Longitudinal (A-14)

b. Thrust

Engine thrust at the four thrust mount attachment points, LC-1 through LC-4

c. Pressures

Engine Oxidizer Injector Pressure

Engine Thrust Chamber Pressure

5.6.2 Test Results

Test results are presented in Tables 5-13 and 5-14 and discussed in the following. Table 5-13 is a summary of all of the vibration, force, and pressure data obtained during engine acceptance testing. Model XLR 81-BA-13 engines (803, 806, and 807) were acceptance tested early in the Sure Fire Program, therefore, some of the accelerometer data (accelerometer nos. 2 through 5, and 8, and 9) were not acquired during the initial tests. The accelerometer data in Table 5-13 shows the peak g's recorded at thrust chamber ignition. Using accelerometer A1 (mounted on the thrust chamber lug) as a reference accelerometer, the following minimum and maximum peak g's were noted during YLR 81-BA-11 and XLR 81-BA-13 engine firings:

- YLR 81-BA-11 engines: Minimum – 8 g's, Maximum – 37 g's
- XLR 81-BA-13 engines: Minimum – 14 g's, Maximum – 40 g's

The load cell data at ignition for a nominal run of the Model YLR 81-BA-11 engine was 11,636 pounds compared with a nominal of 11,896 pounds for the Model XLR 81-BA-13 engine. The load cell data for YLR 81-BA-11 engine 708 was not considered valid

due to load cell calibration inaccuracies. The thrust chamber ignition pressure (Pc) measurements of the YLR 81-BA-11 and XLR 81-BA-13 engines agreed quite well with no significant differences. The oxidizer manifold pressure (OMP) was saturated during YLR 81-BA-11 engine firings and a valid comparison was not possible, however, the XLR 81-BA-13 engine OMP transient data evaluation indicate no adverse pressure peaks.

Table 5-14 shows the results of the Power Spectral Density (PSD) data. The data shows a spread of 0.7 g (RMS) measured at the test rig to 28 g (RMS) at the thrust mount actuator-side. Figures 5-21 through 5-25 are typical PSD plots of the YLR 81-BA-11 and XLR 81-BA-13 engine firings indicated below:

Fig. 5-21 Run No. 435 XLR 81-BA-13 engine 803
Fig. 5-22 Run No. 436 XLR 81-BA-13 engine 806
Fig. 5-23 Run No. 438 YLR 81-BA-11 engine 707
Fig. 5-24 Run No. 439 YLR 81-BA-11 engine 708
Fig. 5-25 Run No. 440 YLR 81-BA-11 engine 709

The area under the curves of Figs. 5-21 through 5-25 represent g (RMS) and is equal to:

$$g\ (\mathrm{RMS}) = \left[\int_{f_1}^{f_2} G_o \ df \right]^{1/2}$$

where

G_o = density (g^2/cps)
f = frequency (cps)

The PSD analyses were made approximately 0.4 seconds after thrust chamber ignition. The PSD plots show distinct peak g (RMS) values at frequencies around 900, 1600, and 1800 cps. Most of the energy represented by the area under each of the PSD plots occurs at these predominant frequencies. These frequencies correlate with the oxidizer and fuel pump impeller blade rotational frequencies of 960 cps for the oxidizer pump and 1690 cps for the fuel pump at steady state turbine speed. The predominant 1800 cps frequency point is the first harmonic of the 900 cps frequency.

Figures 5-26 and 5-27 are typical shock spectra plots showing the analysis of the accelerometer data from accelerometer numbers A-11 for engine 707 and A-1 for engine 708. The data was reduced using an amplification factor (Q) equal to ten where the damping (ζ - zeta) would be 0.05 from the following equation:

$$Q = \frac{1}{2\zeta}$$

Damping of 0.05 is considered nominal in the Agena structure; however, sufficient data is not available to determine the damping on the BAC facility thrust stand installation. If any peak response g value is divided by the amplification factor (in this case Q = 10) it would represent the g value for steady state vibration at that particular frequency. Figure 5-27 shows a peak reading of 57.5 g's occurring at 700 cps. This would represent 5.75 g's of steady state vibration. The shock spectra analyses were made during the thrust chamber ignition period. The typical shock spectra plots (Figs. 5-26 and 5-27) for the YLR 81-BA-11 engine show that the peak response g levels at their resonant frequencies are within the test requirements of LMSC 6117D. The dubbed tape data for the XLR 81-BA-13 engine firings were not adequate for shock spectra analysis. A dynamic analysis of the data in Table 5-13 indicates that shock spectra for the XLR 81-BA-13 engine are similar to the YLR 81-BA-11 engine, with the possible exception in frequency content, due to the difference in engine masses.

5.6.3 Conclusions

The data contained in Tables 5-13 and 5-14 do not indicate any significant difference in engine start transient shock severity between the YLR 81-BA-11 and XLR 81-BA-13 engines when fired at ground level.

The Table 5-13 levels were reduced during the main engine ignition period and, therefore, generated the maximum g levels during each firing. The shock transient levels obtained during the hard-mounted engine tests at BAC compare favorably with the 40g, 8 millisecond test requirements of LMSC 6117D for Agena vehicle component installations.

One run of the 21 engine acceptance tests produced a peak g level of 49 for a duration of approximately 3 milliseconds. The average peak acceleration levels at ignition were less than 40 g's on the remaining 20 tests. Evaluation of the steady state vibration levels indicates that the levels were lower than the test levels specified in LMSC 6117D. During the flight operation the modified, oxidizer-lead, multistart engine is not expected to generate ignition shock levels that are significantly different in magnitude and character from the standard YLR 81-BA-11 engine.

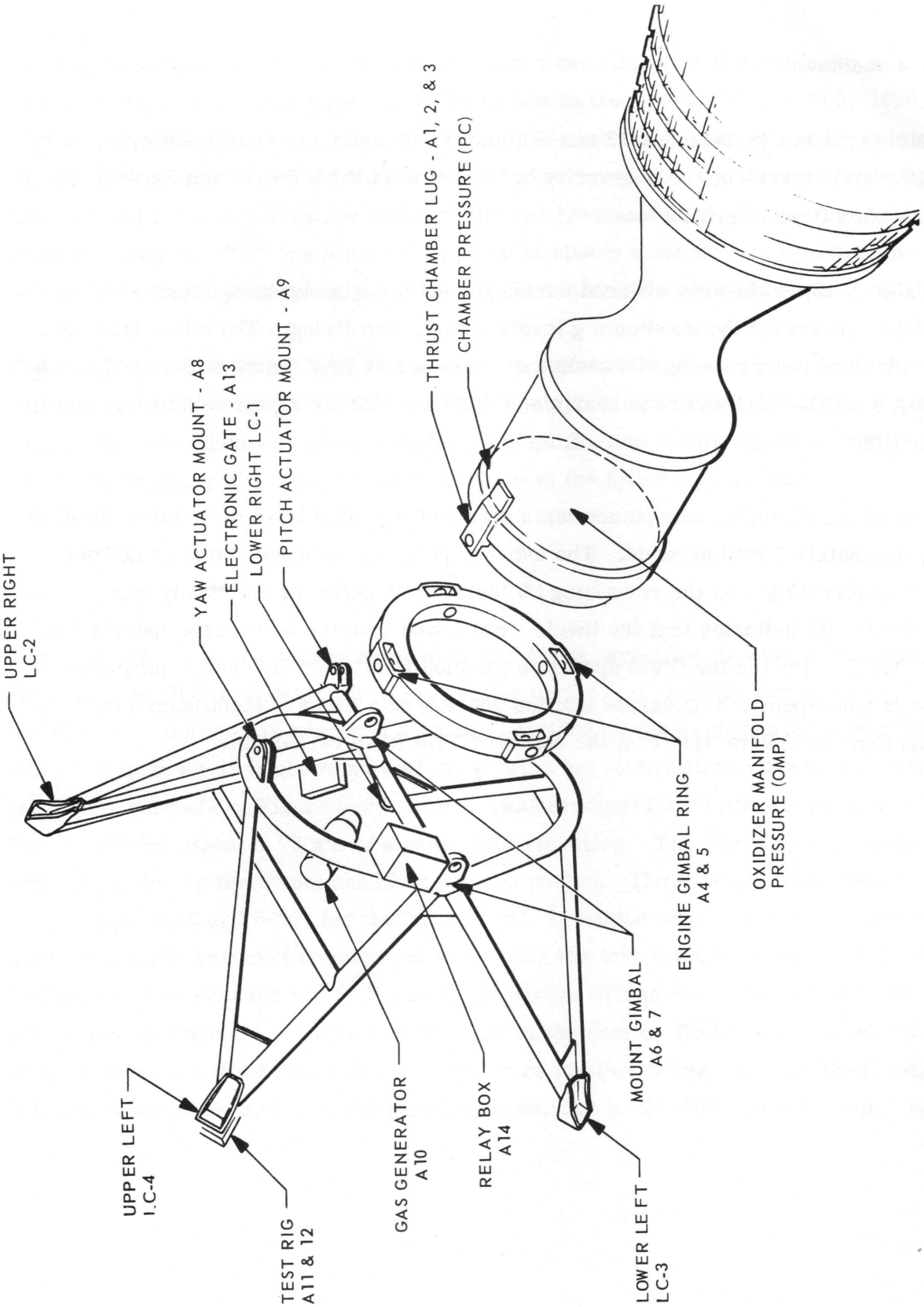
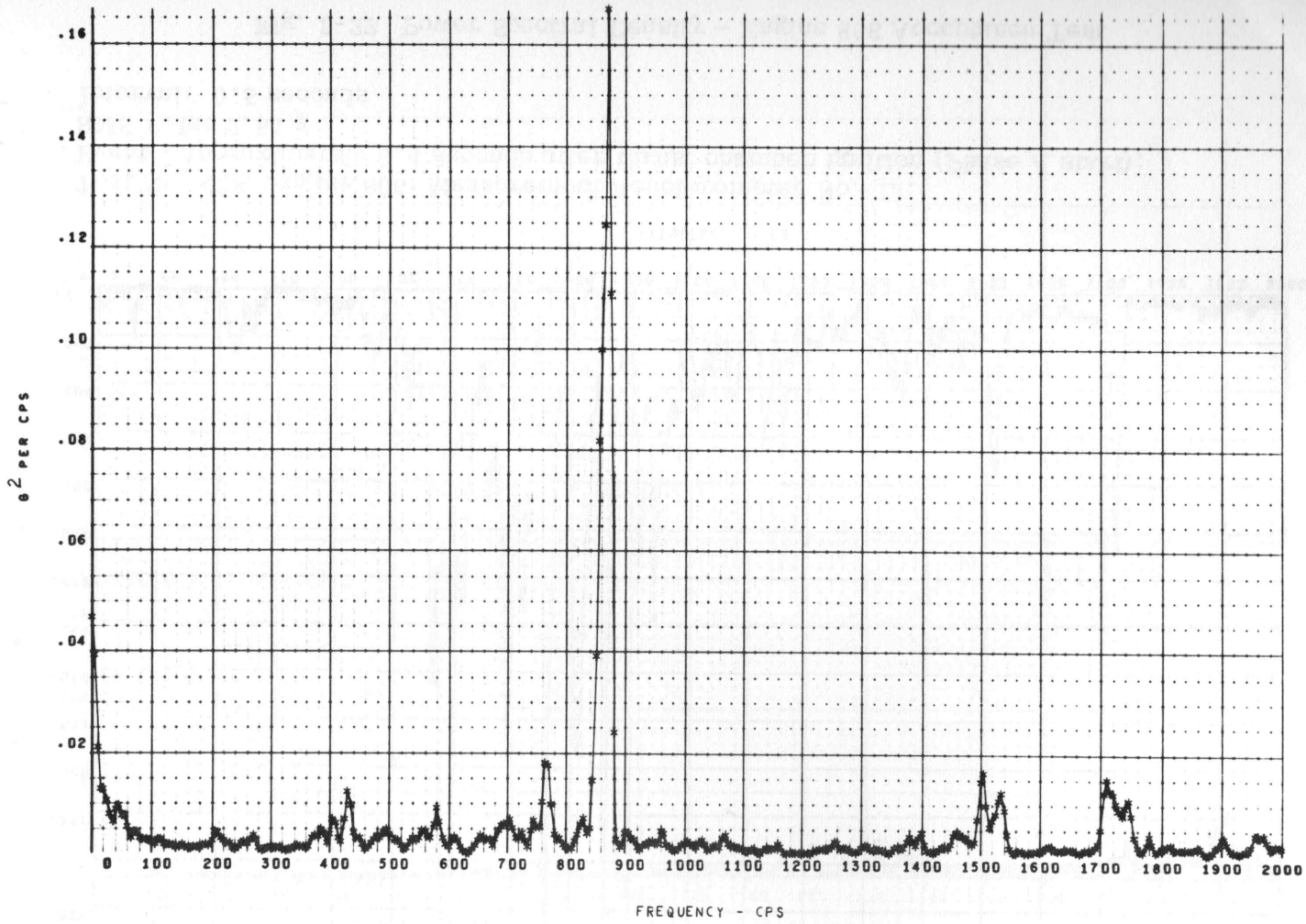
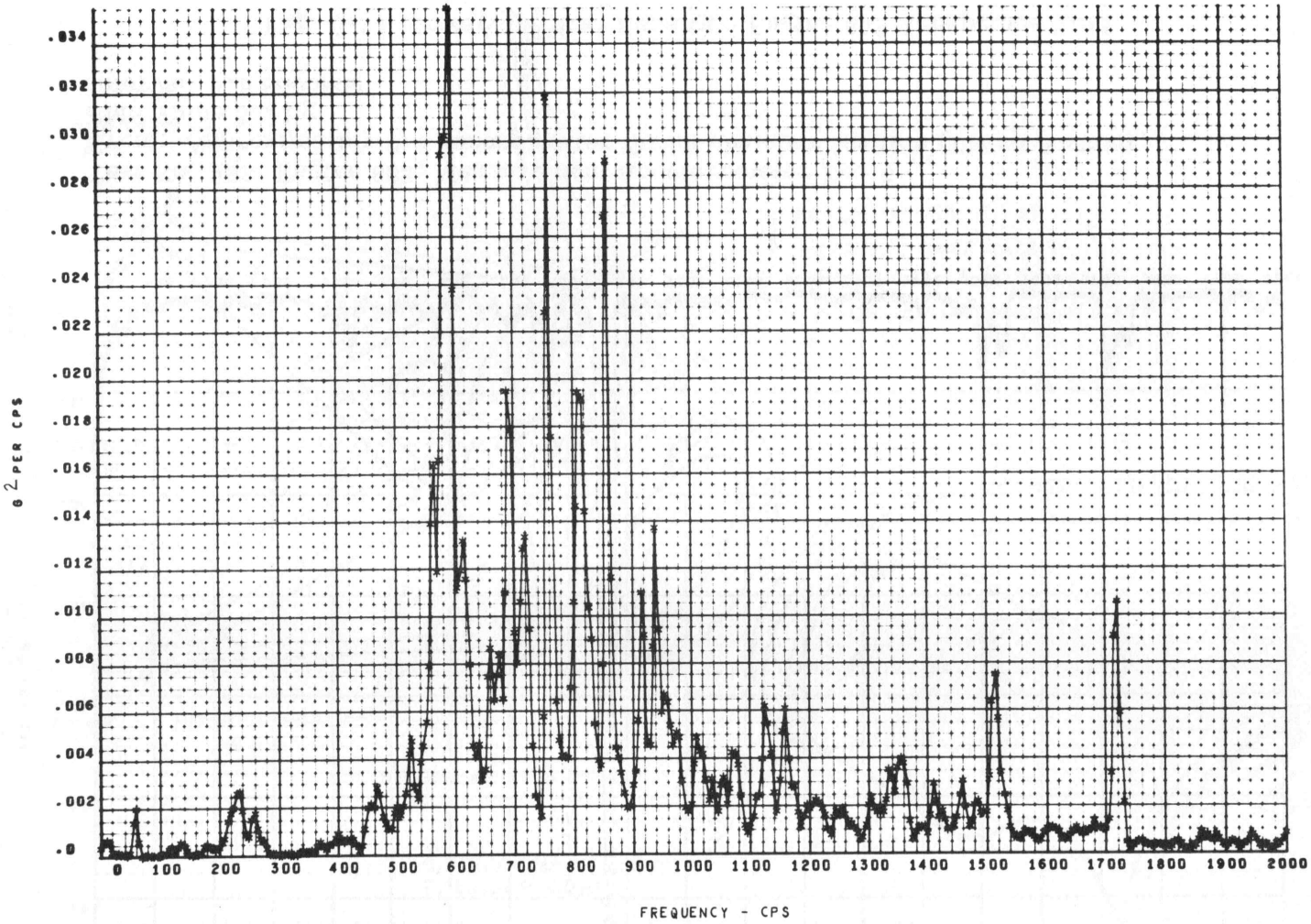


Fig. 5-20 Dynamic Measurement Locations



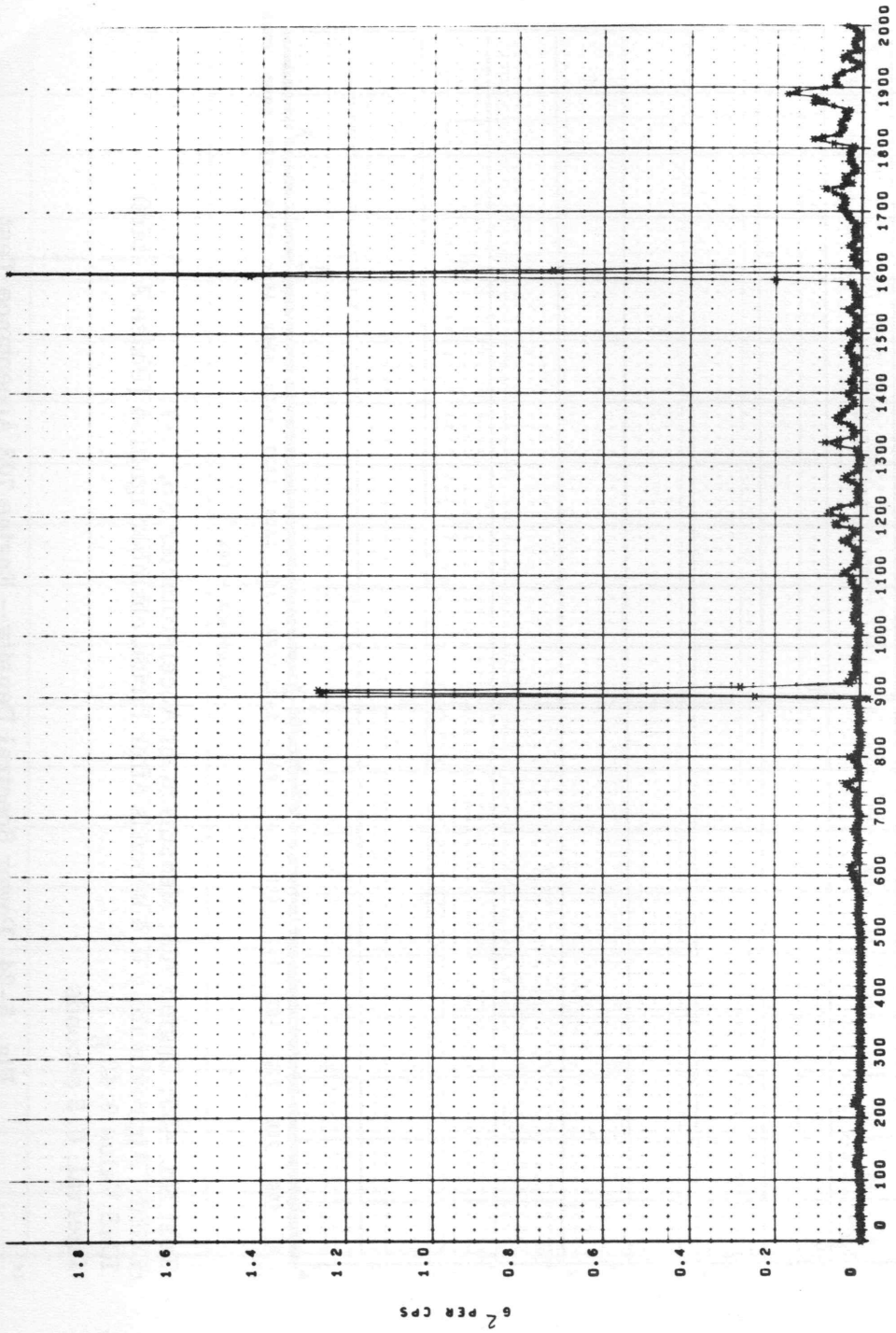
Test No. 435; Engine 803; Measurement Accelerometer No. 6;
Event: Approximately 0.4 seconds after thrust chamber ignition (Phase A start);
RMS Value 3.21 g
Interval: 0.5 Seconds

Fig. 5-21 Power Spectral Density – Engine 803 Acceptance Test



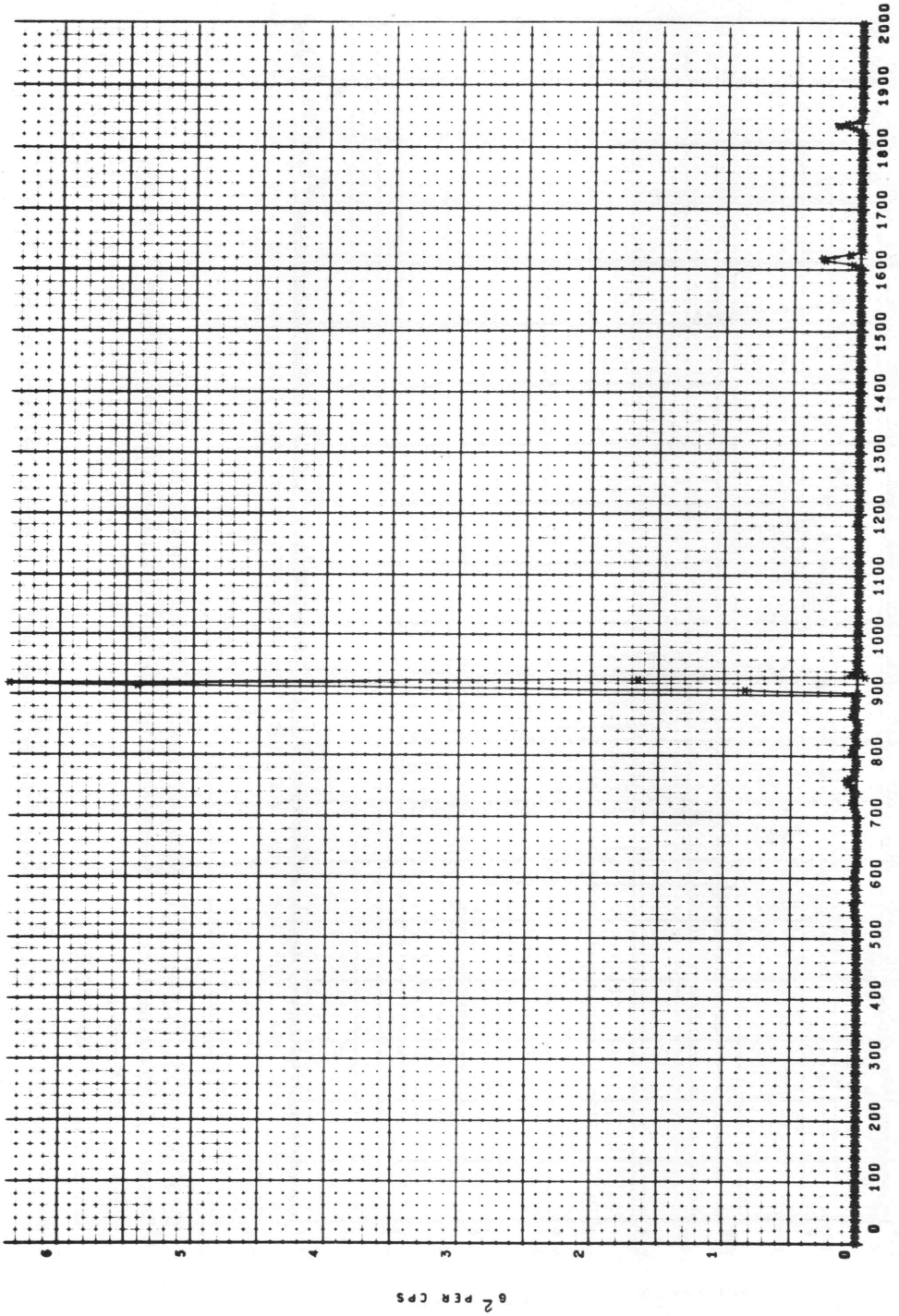
Test No. 436; Engine 806; Measurement Accelerometer No. 13;
Event: Approximately 0.4 seconds after thrust chamber ignition (Phase A start);
RMS Value 2.81 g
Interval: 0.5 seconds

Fig. 5-22 Power Spectral Density – Engine 806 Acceptance Test



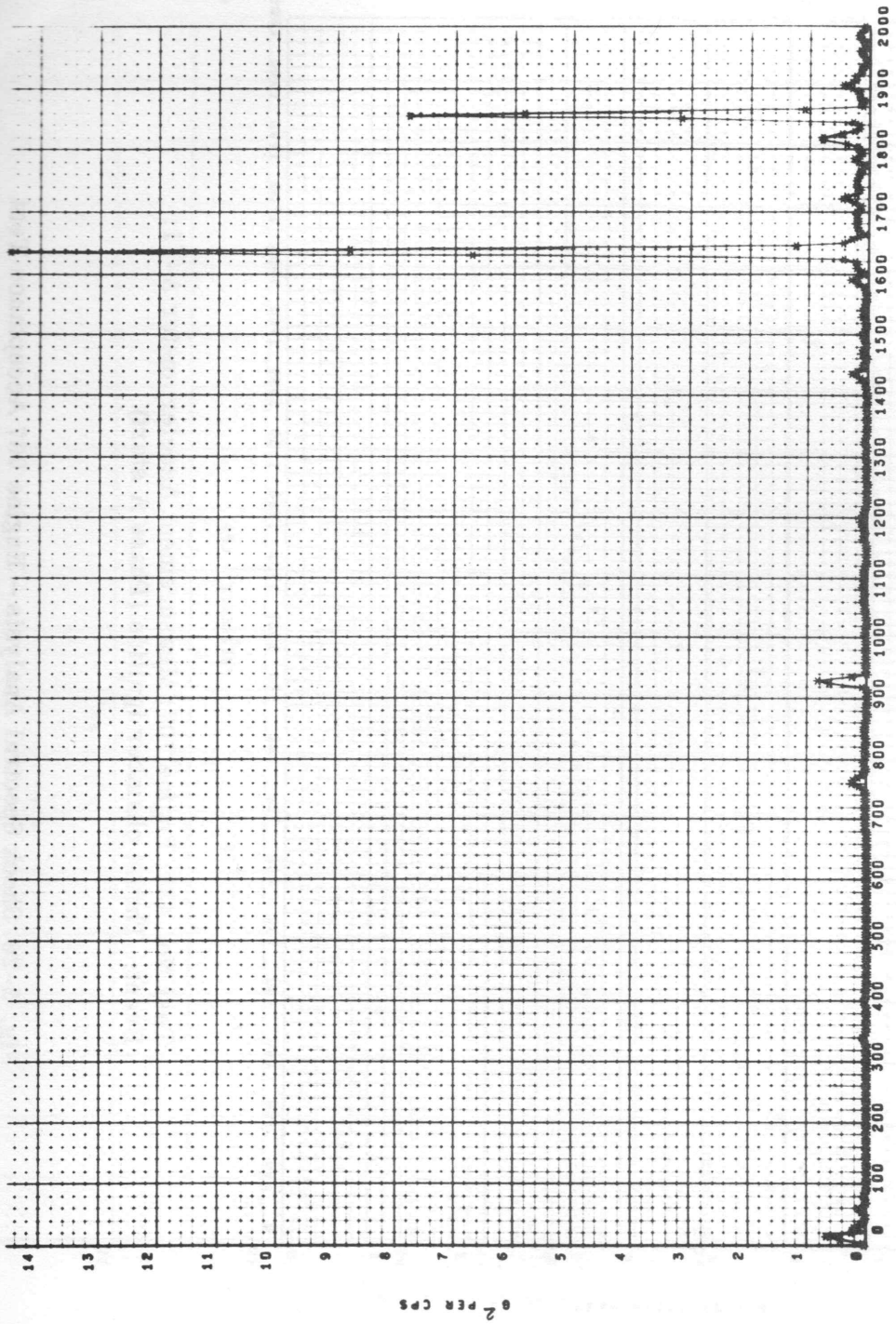
Test No. 438; Engine 707; Measurement Accelerometer No. A-1;
Event: Approximately 0.4 seconds after thrust chamber ignition (Phase A start);
RMS Value 8.72 g
Interval: 0.5 seconds

Fig. 5-23 Power Spectral Density – Engine 707 Acceptance Test



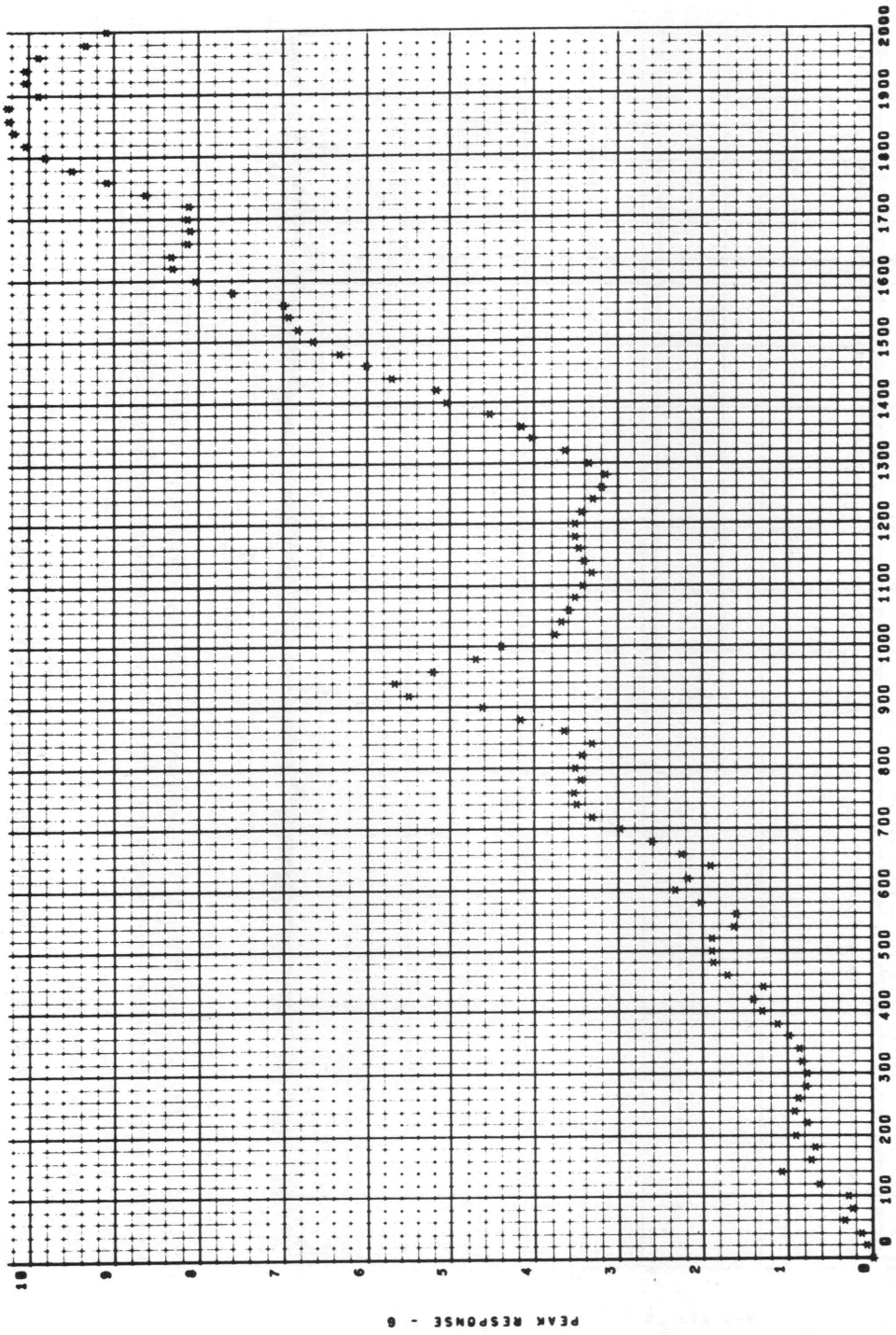
Test No. 439; Engine 708; Measurement Accelerometer No. A-7;
Event: Approximately 0.4 seconds after thrust chamber ignition (Phase A start)
RMS Value 9.40 g
Interval: 0.5 seconds

Fig. 5-24 Power Spectral Density – Engine 708 Acceptance Test



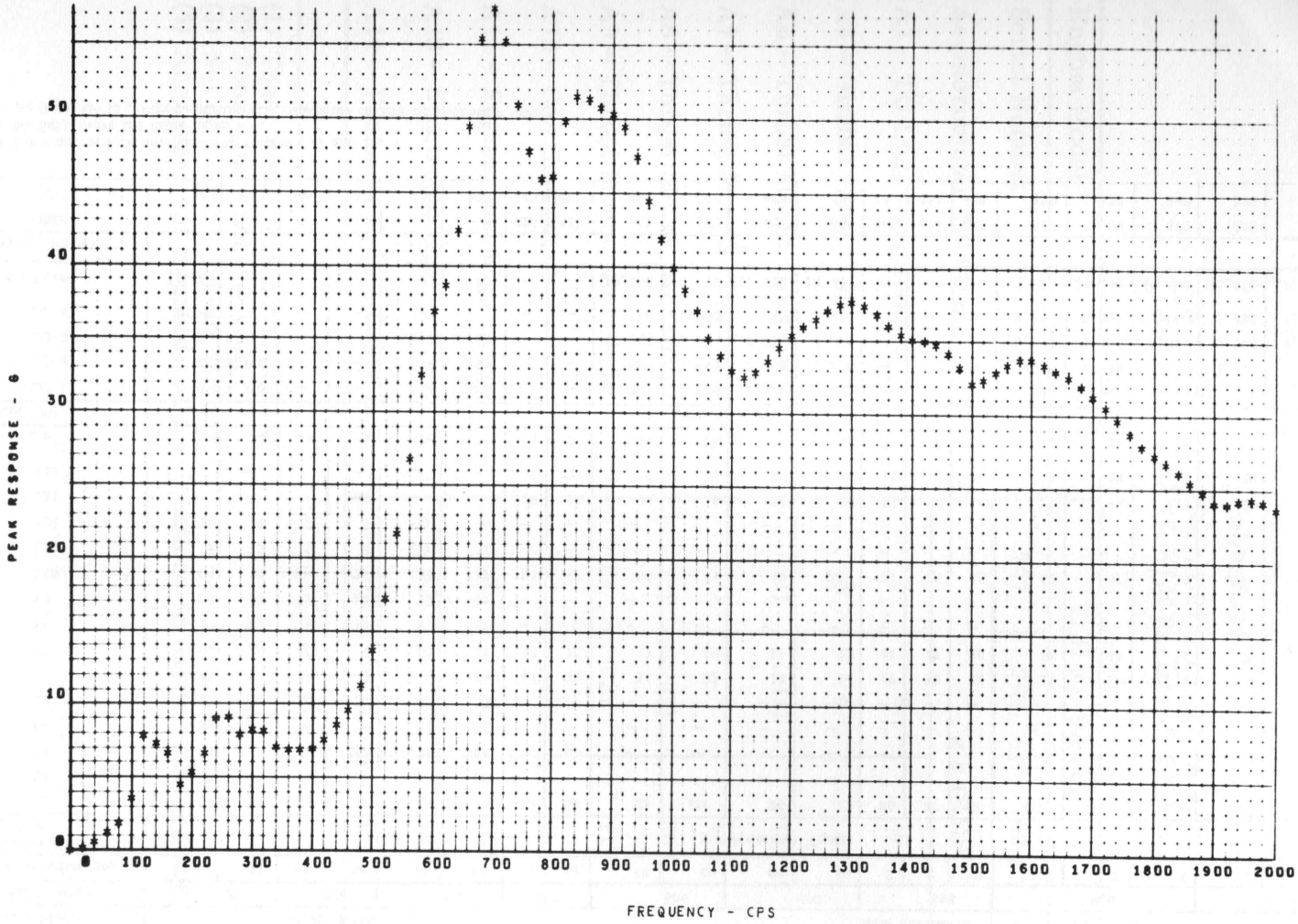
Test No. 440; Engine 709; Measurement Accelerometer No. A-8;
Event: Approximately 0.4 seconds after thrust chamber ignition (Phase A start);
RMS Value 19.91 g
Interval: 0.5 seconds

Fig. 5-25 Power Spectral Density – Engine 709 Acceptance Test



Test No. 438; Engine 707; Measurement Accelerometer A-11;
Event: Thrust chamber ignition (Phase A start)

Fig. 5-26 Shock Spectral Analysis – Engine 707 Acceptance Test



Test No. 439; Engine 708; Measurement Accelerometer A-1;
Event: Thrust chamber ignition (Phase A start)

Fig. 5-27 Shock Spectral Analysis – Engine 708 Acceptance Test

Table 5-13
ENGINE TEST DYNAMIC DATA

Engine Model	YLR 81-BA-11						XLR 81-BA-13														
Serial Number	707		708		709		803		806		807		803			806			807		
Run Duration (sec.)	60	60	60	60	60	60	60	60	60	60	60	60	2	2	2	2	2	2	2	2	2
Accelerometer Number	g's Vector (0-Peak)																				
A1	37	21	28	8	28	17	25	34	22	(2)	20	23	30	37	40	(2)	(2)	(2)	14	19	22
A2	40	13	9	17	13	13	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
A3	21	9	7	(2)	15	10	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
A4	28	6	18	8	35	21	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
A5	22	10	26	10	37	19	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
A6	44	14	13	6	17	15	14	18	14	21	17	14	17	19	17	11	18	14	8	13	9
A7	61	15	17	8	21	23	17	15	11	10	21	12	14	17	16	11	14	14	13	12	10
A8	81	21	16	13	22	17	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
A9	81	53	32	13	17	15	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
A10	94	64	(2)	(2)	153	113	54	68	34	74	49	(2)	51	78	55	42	37	36	(2)	(2)	(2)
A11	8	2	9	9	(2)	7	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
A12	13	3	3	3	(2)	5	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
A13	–	–	–	–	–	–	18	11	19	18	23	16	15	17	14	17	16	15	14	17	19
A14	–	–	–	–	–	–	15	17	9	15	30	27	15	15	14	8	13	13	20	28	30
Ave. g's (1)	49	23	18	9	36	26	24	27	18	28	27	18	24	30	26	18	20	18	14	18	18
Load Cell	Pounds Force																				
LC-1	2,078	1,075	–	–	1,515	1,522	–	–	2,250	2,300	2,620	2,735	–	–	–	2,085	2,360	2,140	2,280	2,400	2,320
LC-2	3,631	2,500	–	–	1,698	2,280	–	–	2,690	3,040	2,695	3,310	–	–	–	2,900	2,810	2,850	2,710	3,180	3,001
LC-3	3,682	2,535	–	–	1,551	1,940	–	–	2,865	2,910	3,250	3,552	–	–	–	3,070	2,870	2,910	3,220	3,425	3,220
LC-4	2,245	–	–	–	–	–	–	–	2,560	2,600	2,630	3,820	–	–	–	2,475	2,220	2,600	3,285	3,385	3,210
Total Force	11,636	–	–	–	–	–	–	–	10,365	10,850	11,195	13,417	–	–	–	10,530	10,260	10,500	11,495	12,390	11,751
Press. Xducer	Psia																				
OMP	–	–	–	–	–	–	625	665	532	562	627	560	561	674	657	582	606	577	648	606	506
PC	321	321	468	355	453	349	440	424	484	447	425	450	407	424	433	439	453	442	375	425	375

(1) Does not include accelerometers Nos. 11 and 12.
(2) No data above the noise level
– No data due to instrumentation discrepancies or low acceleration levels.

Table 5-14

POWER SPECTRAL DENSITY DATA FOR
YLR 81-BA-11 AND XLR 81-BA-13 ENGINES

Engine Model	YLR 81-BA-11			XLR 81-BA-13	
Serial Number	707	708	709	803	806
Accelerometer Locations	g's (RMS)				
A1 – TC Lug, Longitudinal	9	8	10	(2)	(2)
A4 – Gimbal Ring, Top	6	4	3	(4)	(4)
A6 – Thrust Mount, Left Gimbal	(1)	(1)	(1)	3	3
A7 – Thrust Mount, Right Gimbal	(1)	9	11	2	3
A8 – Thrust Mount, Act. Top	(1)	(1)	20	(4)	(4)
A9 – Thrust Mount, Act. Side	28	22	(1)	(4)	(4)
A10 – Gas Generator	17	(2)	25	6	8
A11 – Engine Test Rig, Long.	7	6	(1)	(4)	(4)
A13 – Electronic Gate	(3)	(3)	(3)	8	3
A14 – Pressure Switch Relay Box	(3)	(3)	(3)	3	3

(1) These acc. were not analyzed.
(2) Poor data – not analyzed
(3) Not on YLR 81-BA-11 engines.
(4) Acc. were not mounted at these locations on engines 803 and 806.

5.7 OXIDIZER FEED PRESSURE SWITCH DURABILITY TEST PROGRAM

Design changes during redevelopment of the XLR 81-BA-13 engine resulted in addition of an oxidizer manifold pressure switch (OMPS), oxidizer feed pressure switch (OFPS), and associated junction boxes for actuation of the pilot operated solenoid valve (POSV). The selection of an OFPS as a redundant start signal source was based on the successful application of such a switch on the Ranger series YLR 81-BA -9 engines. Early in the turbine pump test program it became apparent that the turbine pump start transient associated with the XLR 81-BA-13 liquid-start gas generator system was different than that associated with the solid-propellant start system of the YLR 81-BA -9 engine. The major differences affecting the use of the OFPS was a higher mean pressure at the OFPS port on the main oxidizer valve prior to thrust chamber ignition. This higher mean pressure together with normal pressure perturbations resulted in a rapid actuation-deactivation cycling of the OFPS which had not been considered in the original design of the switch application.

The turbine pump test program was modified to investigate the effects of this cyclic operation on the OFPS durability and to investigate the possible damping effect of an orificed OFPS sensing line. A total of 66 test runs were made as shown in Table 5-15.

5.7.1 Test Configuration

The test setup used for normal turbine pump testing (paragraph 5.4) was utilized with minor changes. These changes were to:

a. Resize the gas generator venturis to control turbine speed.
b. Add an oxidizer discharge line throttle valve to control OFPS cycling at the maximum rate throughout the run.
c. Provide additional instrumentation to record switch output and POSV actuation and intermediate stage pressures.

5.7.2 Discussion

The basic intent of the OFPS durability tests was to evaluate the mechanical durability of the switch particularly the microswitch spring fatigue limits. Switching rates as high as 250 cps, and pressure variations in the OFPS line as high as 1000 cps had been observed during turbine pump testing. Phase differences up to 180 degrees occurred between pressure variation and switch position. These differences imposed high dynamic loads on the mechanical actuation mechanism of the microswitch within the pressure switch assembly (Fig. 5-28).

During a normal engine start, with the OMPS commanding the POSV, the OFPS would cycle 20 to 30 times. Four or five of these cycles would be inductive (POSV current) load carrying cycles prior to OMPS actuation. During an engine start wherein the OFPS was commanding the POSV (an OMPS malfunction), the OFPS would cycle 25 to 50 times prior to engine thrust development. Of these cycles, 20 to 35 would carry POSV current load prior to thrust chamber ignition.

Subsequent back pressure in the oxidizer line raised the RMS pressure value well above the OFPS actuation pressure range. Thus, evaluation of continuous running in the cyclic mode was not of particular concern; however, the accumulative results of multiple starts could exceed the switch mechanical life design margin. The tests conducted, with one exception, showed no mechanical failure or degradation of the microswitch actuation mechanism in a total of 56 runs on six switches, comprising approximately 60,000 cycles of switch operation. The exception occurred in test 2101 in which switch number A10120 was run for 240 seconds, comprising also approximately 60,000 cycles. Prior to this test the same switch had, in test 2100, been tested over 130 seconds total. At the conclusion of test 2101, the switch was disassembled and the beryllium-copper leaf-spring was found to be significantly deformed as shown in Fig. 5-29. To obtain this nature of deformation, a spring temperature in excess of 800° F was required. The other five switches (tests 2095 through 2099) showed no mechanical failures or degradation of the microswitch actuation mechanism.

Tests which imposed a 2.4 to 3.0 ampere POSV inductive load caused severe burning of the OFPS electrical contacts. In two tests, however, 1.2 ampere resistive loads were imposed, and no contact degradation was found. In tests which used a POSV inductive load, the anode contact (movable spring contact) was pitted and the cathode contact (normally-open contact) had gross metallic-oxide buildup. Tests were run with and without the EMI suppression network but no difference in the severe contact erosion occurred when the inductive load was used.

Extended cycling of the switch at high-inductive current loads caused destruction of the silver contacts and resulted in severe heat buildup in the Be-Cu spring over extended operating times. The test conditions under which this occurred are 50 to 100 times more severe than normal engine operation in terms of number of load carrying cycles of operating.

5.7.3 Conclusions

The mechanical durability of the OFPS and its incorporated microswitch exceeded the limits of the testing accomplished. Imposition of inductive electrical loads on the switches during testing resulted in severe heat buildup during the tests and severe degradation of the electrical contacts. Normal engine operating conditions result in a significantly lower number of cycles (20 to 30) of switch operation than those used in the testing. Results of the vacuum testing (paragraph 5.8) and altitude testing (Section 7) showed no degradation of either mechanical or electrical properties when the switches were operated at conditions, either simulated or actual, encountered in normal engine operation, including start without an OMPS in command.

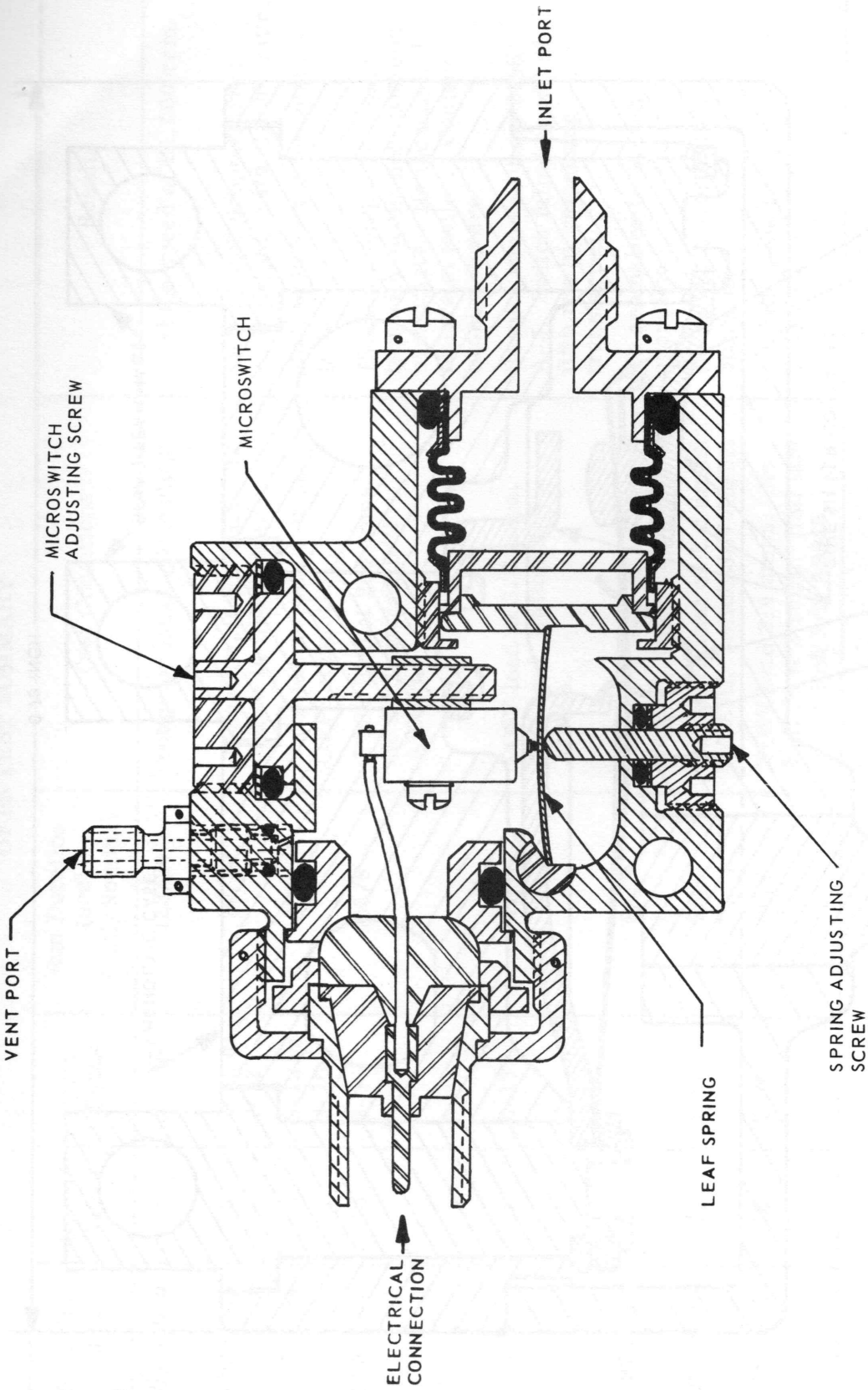


Fig. 5-28 Cross Section of Pressure Switch (Typical of Oxidizer Manifold, and Feed Pressure Switches)

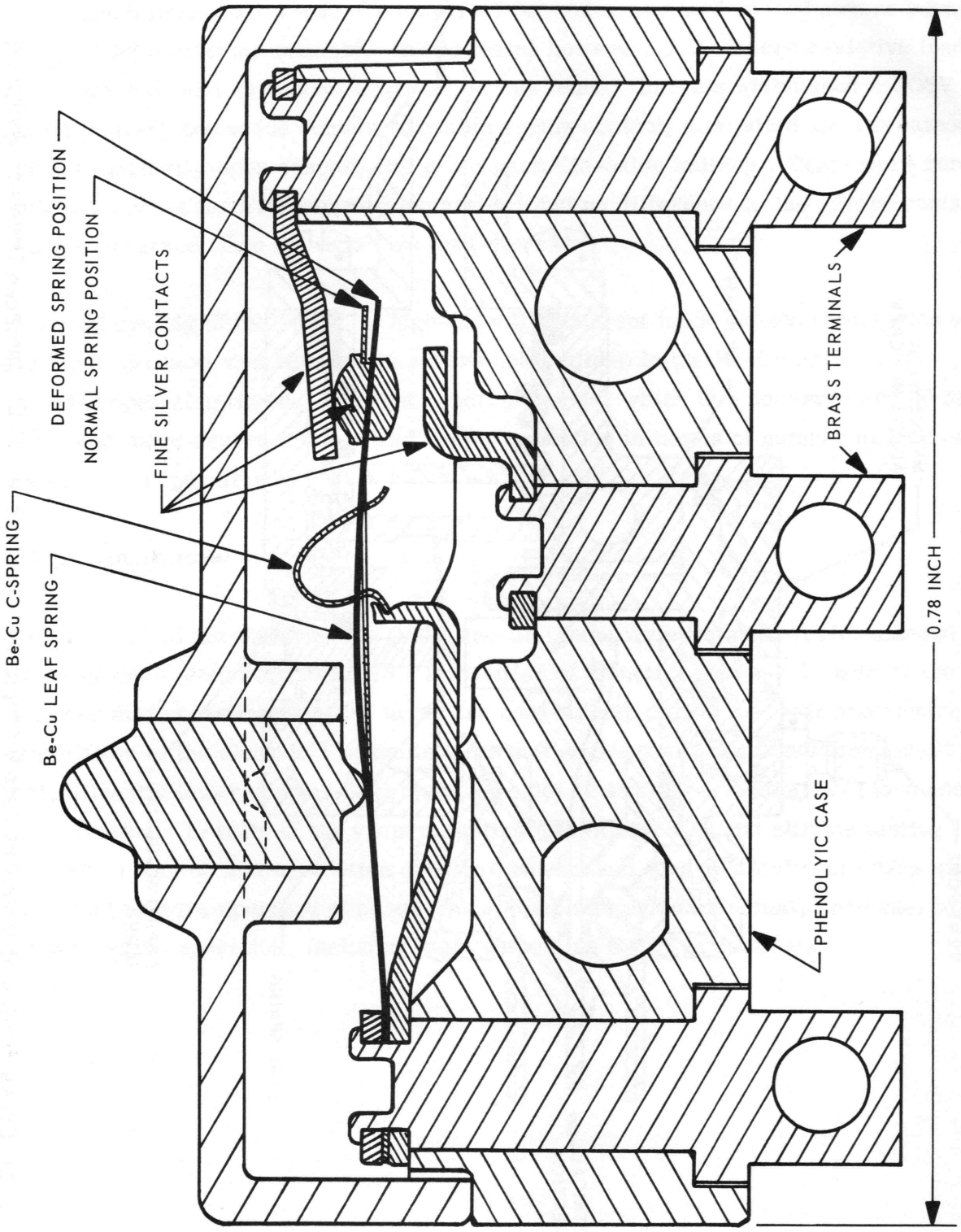


Fig. 5-29 15M179 Microswitch (Showing Position of Deformed Spring)

Table 5-15

OFPS TEST SUMMARY

BAC Run Number	No. of Starts	Run Dates	Run Duration (sec) Total/Nominal	Test Conditions	Results
2086-2089	4	2-1 to 2-9	123/30	Setup for max. ΔP cycling.	Turbine speed of 21,000 rpm.
2090-2092	3	2-11 to 2-12	90/30	No orifice, 0.050 and 0.030 orifice in OFPS line.	P-P cycling dropped from 62 to 47 psig with 0.030 orifice.
2093-2094	2	2-15 to 2-16	60/30	Setup for max chatter on OFPS contacts.	R&D switch used for setup failed.
2095 A-J	10	2-16	105/8	POSV load at 2.4 amps. No EMI, switch S/N A10585.	Electrical output not consistant with mechanical position of contacts. No mechanical failure of switch. Contact erosion severe.
2096 A-E	5	2-18	41/8	POSV load at 3.0 amps. EMI used, switch S/N A10588.	Results same as runs 2095 A-J. Switch failed at end of fifth run by remaining transferred.
2097 A-J	10	2-19	83/8	Resistive load of 1.2 amps. No EMI, Switch S/N A10590.	No anomalies in switch operation.
2098 A-K	11	2-19	96/9	Resistive load of 1.2 amps. No EMI, switch S/N A9837.	No anomalies in switch operation.
2099 A-J	10	2-20	84/8	POSV load 2.4 amps. No EMI first 2 runs. EMI last 8. Switch S/N A9945.	Results same as runs 2095 A-J.

LOCKHEED MISSILES & SPACE COMPANY

Table 5-15 (Cont.)

BAC Run Number	No. of Starts	Run Dates	Run Duration (sec) Total/Nominal	Test Conditions	Results
2100 A-J	10	2-20	133/8	POSV load at 2.4 amps. EMI used. POSV hydraulic operation monitored. Switch S/N A10120.	Results same as runs 2095 A-J.
2101	1	2-20	240	Long run degradation test of switch used in run 2100 A-J. Same conditions as run 2100 A-J.	Current through switch decreased with run time. Same results as run 2095 A-J.

5.8 PRESSURE SWITCH VACUUM TESTS

The basic intent of the pressure switch vacuum testing was to evaluate the durability of the microswitch contacts used in the various engine pressure switches under conditions of high vacuum and nominal engine operating cycles. The tests were conducted by Cornell Aero Labs in Buffalo, New York for Bell Aerosystems Co. on 19 through 27 February 1966. The test sequence shown in Table 5-16 was established based on a composite mission profile for the XLR 81-BA-13 engine except that the equivalent number of engine starts was 38.

5.8.1 Test Configuration

The microswitches were mounted in vacuum chamber using a Cornell Aero Lab-supplied actuation mechanism. A solenoid coil from a pilot operated solenoid valve (POSV) with EMI suppression, was supplied by BAC. A test cycle consisted of one operation of an oxidizer manifold pressure switch (OMPS) microswitch followed by 20 operations of an oxidizer feed pressure switch (OFPS) microswitch. The POSV coil was used with each microswitch for each test cycle. Coast times between cycles was as shown in Table 5-16.

5.8.2 Discussion

The microswitch used in both the OMPS and OFPS assemblies is shown in Fig. 5-29. Microswitch contact erosion and tempering of the Be-Cu leaf spring were observed in the mechanical durability tests wherein the microswitches were continually cycled under full load conditions far in excess of normal engine cyclic operation.

The microswitches used in the OMPS and OFPS are manufactured by Minneapolis Honeywell Microswitch Division as a special modification of a standard 1SM3 switch. The modified microswitch is designated 1SM179 and was designed specifically for use in pressure switch assemblies of the BAC engines. The nominal operating and

dimensional characteristics of the 1SM179 microswitch are given in Table 5-17 with those of the standard 1SM3 for comparison. The lower minimum cycle life is a derated figure based upon the greater overtravel and thinner spring material used in the 1SM179 and is a conservative figure. In the mechanical durability tests the switch used in tests 2100 and 2101 mechanically survived more than 60,000 cycles even with the excessive contact erosion and heating caused by the high cyclic rates and inductive loading. The manufacturer's data provides no electrical ratings above 50,000 feet altitude. Previous qualification tests of the pressure switch assembly, and thus the microswitch which is internally mounted, were at an altitude of 300,000 feet. These tests were run with 1.2 ampere inductive loads for a total of 100 cycles. Several LMSC Agena Military programs have used similar M-H microswitches in applications above 800,000 feet. Approximately twenty of the switches listed in Table 5-18 were applications wherein the first operation of the switch was after 10 days of on-orbit environment. Of the total of over 200 switches flown, one possible mechanical malfunction was experienced which was related to mechanical installation of the switch and was not an inherent microswitch malfunction. The maximum current handled, from Table 5-18, was 1.0 ampere inductive which is below the levels required for Gemini application.

No other ground or flight test data was available for high-vacuum application at high-inductive loads under cyclic operation. The tests performed at the Cornell Aero Labs provided sufficient data on this mode of operation to warrant continued use of the engine electrical control system design utilizing the OMPS and OFPS combination. Thirty-eight engine start sequences were run which resulted in 760 cycles of OFPS microswitch operation under full load. AEDC test data (CB series) showed an average of five OFPS full cycles (load carrying) prior to OMPS make for a normal OMPS start and an average of 27 OFPS full cycles for an OFPS start. The total of 33 OMPS and 5 OFPS starts were made on the AEDC switches prior to disassembly. Examination of these switches confirmed the results of the Cornell Aero Lab tests for an altitude range of 300,000 to 400,000 feet. Forty-three OMPS and one OFPS starts were accomplished at sea level conditions during the engine flightworthiness demonstrations and subsequent examination showed no degradation of the microswitch contacts.

5.8.3 Conclusions

No anomalous operation was found in any test. The microswitches were disassembled and all contact surfaces were examined; no evidence of cold welding or contact degradation was observed.

Normal engine operating conditions during either OMPS or OFPS start command have no detrimental effects on the microswitch contacts over the altitude range of the planned Gemini missions.

Table 5-16

VACUUM SWITCH TEST DATA

Test Cycle Number	Date	Coast Time Since Prior Test Cycle	Test Altitude	Pressure (MM Hg)
1	2/20/66	(Start of Testing)	2,000,000 ft	3×10^{-9}
2	2/21/66	24 hr	2,000,000 ft	3×10^{-9}
3, 9, 15, 21, 27	See Note 1	16 hr (Note 2)	2,000,000 ft	3×10^{-9}
4, 10, 16, 22, 28	↓	1.25 hr	2,000,000 ft	3×10^{-9}
5, 11, 17, 23, 29		1.25 hr	2,000,000 ft	3×10^{-9}
6, 12, 18, 24, 30		1.25 hr	2,000,000 ft	3×10^{-9}
7, 13, 19, 25, 31		1.25 hr	2,000,000 ft	3×10^{-9}
8, 14, 20, 26, 32	See Note 1	1.25 hr	2,000,000 ft	3×10^{-9}
33	2/27/66	16 hr	1,400,000 ft	3×10^{-8}
34	2/27/66	1 hr	900,000 ft	3×10^{-7}
35	2/27/66	1 hr	525,000 ft	3×10^{-6}
36	2/27/66	1 hr	367,000 ft	3×10^{-5}
37	2/27/66	1 hr	317,000 ft	3×10^{-4}
38	2/27/66	1 hr	277,000 ft	3×10^{-3}

NOTES: 1. Tests 3 to 8 on 2/22, 9 to 14 on 2/23, 15 to 20 on 2/24, 21 to 26 on 2/25, 27 to 32 on 2/26.
2. Except coast time prior to test 3 was 20 hours.

Table 5-17

MANUFACTURER MICROSWITCH SPECIFICATIONS

	(1SM3) Standard	(1SM179) Modified (1SM3)
Actuation Pressure	3–5 oz.	3 oz. max.
Pre travel	0.030 in.	0.030 in.
Differential Travel	0.004 in.	0.002 in.
Over Travel	0.005 in.	0.010 in.
C-Spring Thickness	0.0030 in.	0.0025 in.
Minimum Cycle Life	200,000	10,000
Contact Material	Fine Silver	Fine Silver
Leaf Spring Material	Be-Cu	Be-Cu
C-Spring Material	Be-Cu	Be-Cu
Electrical Ratings	(Same for 1SM3 and 1SM179)	
	5 amps Resistive at Sea Level	
	5 amps Resistive at 50,000 ft	
	3 amps Inductive at Sea Level	
	2.5 amps Inductive at 50,000 ft	
	5 amps at 250 vac U. L. Rating	

Table 5-18

MICROSWITCH FLIGHT USAGE

Switch Number	Load (Milliamps)	Load Type	Number Flown	Mission Time	Approx. Number of Operations Per Mission – Ground/Flight
1SX-1	10–50	Resistive			
1SM-1	10–50	Resistive	Approx. 200	10 days	40,000/5,000 (See Note 1)
11SM-1	10–50	Resistive			
V3	500–1000	Inductive	Approx. 40	10 days	(Not reported)

NOTE: 1. Approximately 20 switches required only one operation subsequent to 10 days on-orbit with a 2–3 milliamp load.

Section 6
SEA LEVEL FLIGHTWORTHINESS DEMONSTRATION TESTS

6.1 OVERALL OBJECTIVES

The sea level flightworthiness demonstration tests entailed component level testing of the pressure switch relay box, shock and vibration testing of the modified engine hardware installation, and a series of complete engine assembly test firings at sea level conditions.

Upon completion of the sea level tests, test engine 803 was used in the simulated altitude tests conducted at AEDC.

6.2 PRESSURE SWITCH RELAY BOX

The pressure switch relay box was a new component developed for the XLR 81-BA-13 engine Sure Fire modification. The circuit design, functional requirements, and packaging were new, thus, qualification by similarity to other BAC-developed relay boxes was not possible. Flightworthiness tests of the relay box were undertaken to demonstrate compliance with the applicable test and environmental requirements of LMSC specification 6117D and LMSC engine model specification 1414463. Testing was conducted at BAC between 2 February and 24 February 1966.

6.2.1 Test Configuration

Two P/N 8247-472483-1 relay boxes, which were considered representative production units, were used. The boxes were assembled under normal manufacturing and inspection conditions and had successfully passed the acceptance test. A schematic diagram of the test setup used for both boxes is shown in Fig. 6-1. During the dynamic testing, shock mounts were used, a harness cable was routed and clamped similar to the engine frame installation, and "input" and "response" accelerometers were used as shown in Fig. 6-2. The accelerometer outputs and pilot operated solenoid valve (POSV) voltage

were recorded on magnetic tape for high-response analysis. The tests were conducted in accordance with BAC procedure 8247-928308.

6.2.2 Test Results

No anomalies were observed during the testing (see Table 6-1) except under sine vibration operation when the magnetic tape data showed several dropouts on POSV voltage. Examination of data and test instrumentation showed that a test connection was functioning intermittently. Subsequent test runs showed no evidence of dropouts. No visible structural failures were observed in the relay box or shock mounting hardware. All functional tests were passed and no relay chatter was indicated on the magnetic tape data. No insulation breakdowns nor excessive leakage currents were observed during any of the dielectric strength or insulation resistance tests. Operation of the relay box in an explosive atmosphere resulted in no ignition of the atmosphere and internal and external examination revealed no corrosion as a result of the humidity tests.

The pressure switch junction box (refer to paragraph 5.2) is identical in manufacturing methods, packaging design, and mounting and is considered qualified by similarity based on the test results for the POSV relay box.

6.2.3 Conclusion

The pressure switch relay box and pressure switch junction box both are qualified and have adequately demonstrated flightworthy operation during the component level tests.

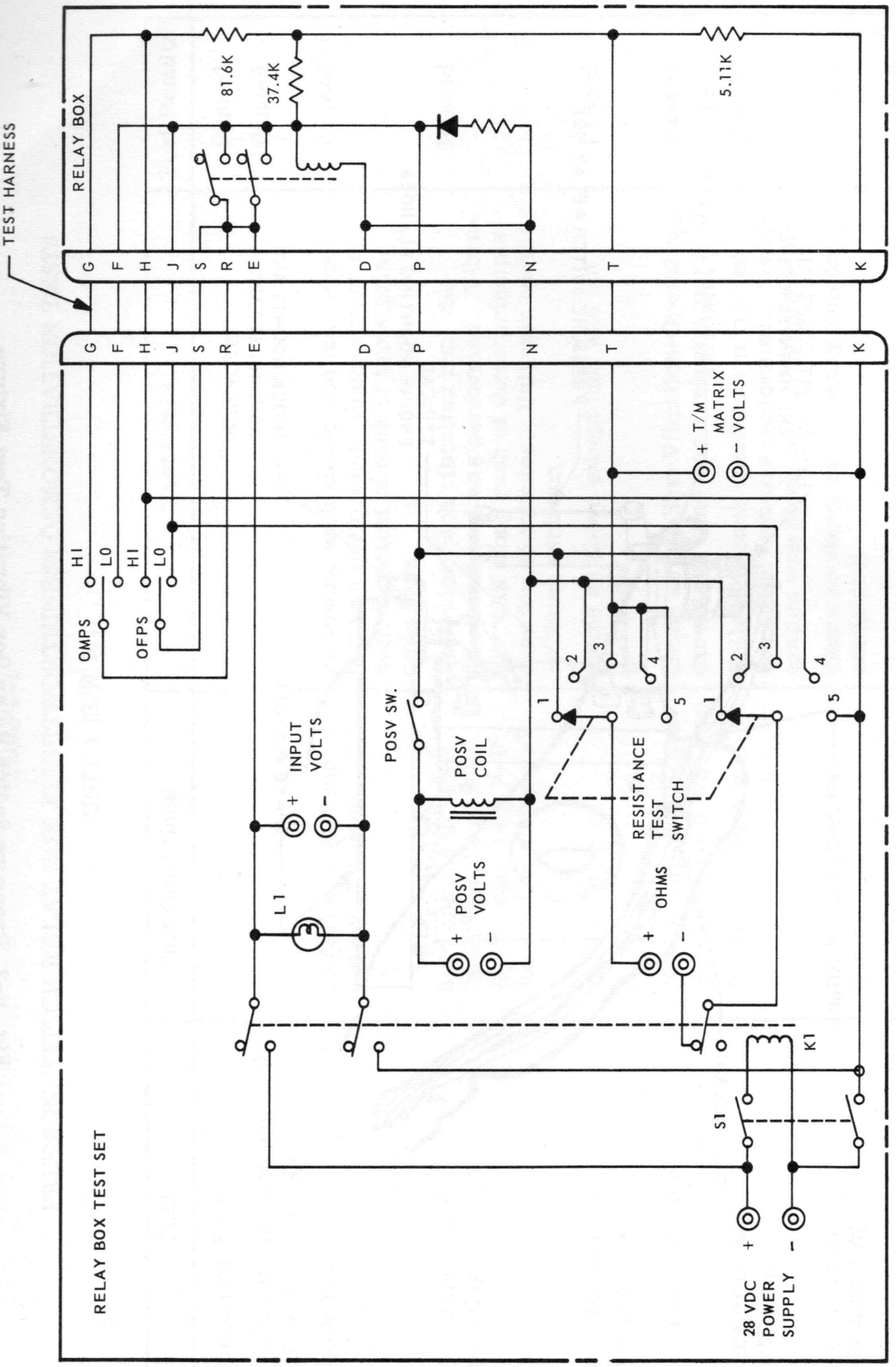


Fig. 6-1 Pressure Switch Relay Box Test Schematic

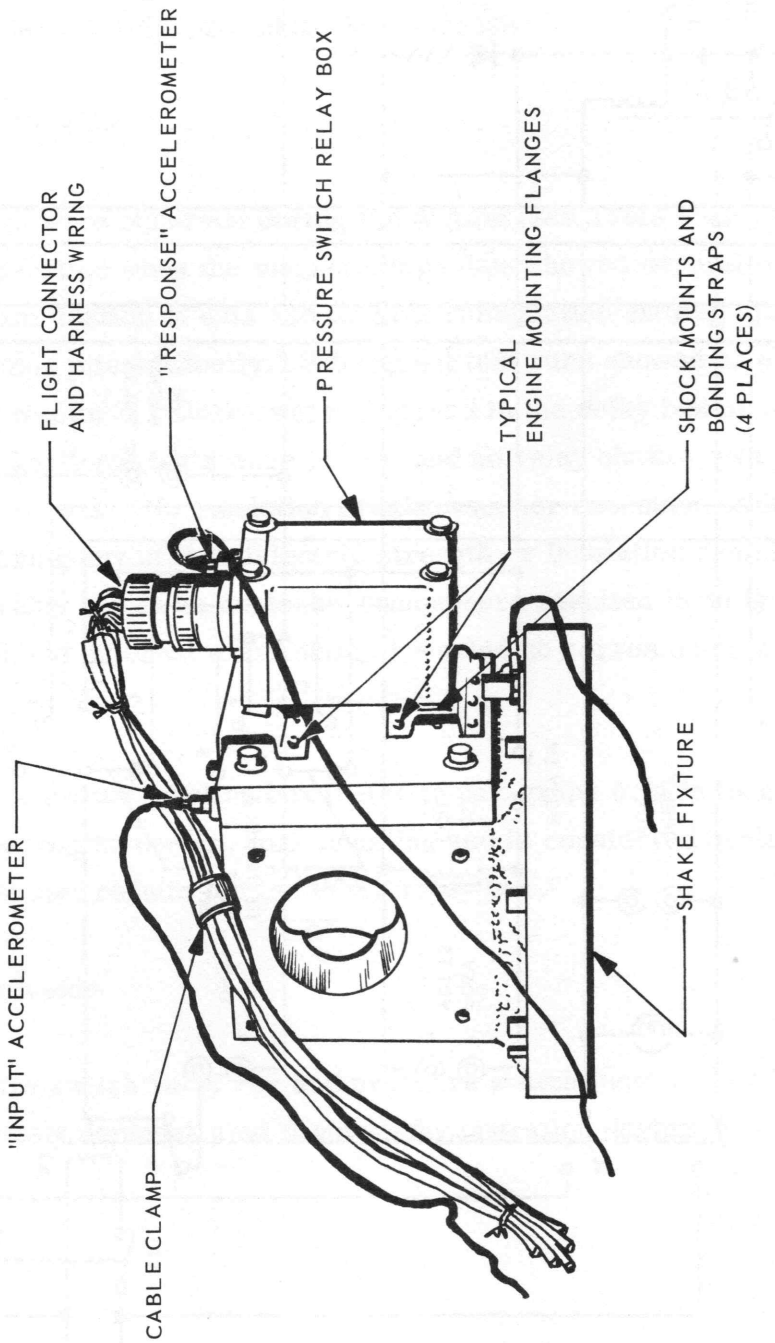


Fig. 6-2 Pressure Switch Relay Box Vibration Test Fixture

Table 6-1

PRESSURE SWITCH RELAY BOX FLIGHTWORTHINESS DEMONSTRATION TESTS

UNIT 1 (S/N 11)

Test	Test Conditions	Test Cycle	Test Results
Functional Test	–	–	Passed
Insulation, Resistance, and Dielectric Strength	–	–	Passed
Shock Test	44 (±4) g, 8 (±1) ms approx. 1/2 sine	Following each shock, the functional test was performed. Shocks were in each direction of each of three principle axes.	Passed
Sine Vibration – Operating	5-14 cps .5 in. P-P 14-40 cps 10 g P-P 40-400 cps 15 g P-P 400-3000 cps 40 g P-P	Following each vibration run, the functional test was performed. Vibration was along each of three principle axes for 25 minutes. Unit was operated 10 times each run.	Passed
Sine Vibration – Nonoperating	Same as above	(Same as above except unit was not operated)	Passed
Random Vibration	20-400 cps .08 g^2/cps 400-2000 cps 0.2 g^2/cps	(Same as for Sine Vibration-Operating, except run time was 5 minutes per axis)	Passed
Acceleration – Operating	12 (±1/2) g for 5 min.	Following each acceleration run the functional test was performed. Acceleration was along each principle axis.	Passed
Acceleration – Nonoperating	15 1/2 (±1/2) g for 5 min.	(Same as above except relay was de-energized)	Passed
Insulation, Resistance, and Dielectric Strength	–	–	Passed
Functional Test	–	–	Passed

Table 6-1 (Cont.)

UNIT 2 (S/N 12)

Test	Test Conditions	Test Cycle	Test Results
Functional Test	–	–	Passed
Insulation, Resistance, and Dielectric Strength	–	–	Passed
Altitude	0.2 psia (95,000 ft.)	Pre-altitude insulation resistance test followed by 4 tests of 25 cycles operation each at 30.5, 19.5, 24, 28 VDC, followed by altitude insulation resistance tests. Ambient functional test, insulation resistance and dielectric strength test at completion of altitude tests.	Passed
Explosion Proof	Sea level, 10, 20, 30, 40, and 50 thousand feet.	Relay box cycled 10 times in explosive atmosphere at each altitude. Ambient insulation and dielectric strength test performed at completion of explosion proof tests.	Passed
Humidity Test	95 percent RH at 100° F	One cycle from 110° F @ 95% RH to 35° F @ 95% RH followed by 5 days at 110° F @ 95% RH. Ambient functional test, insulation resistance and dielectric strength test at completion of humidity tests.	Passed

6.3 ENGINE XRM-4 DYNAMIC TESTS – VIBRATION AND SHOCK

Vibration and shock flightworthiness demonstration tests were conducted to demonstrate the structural integrity of the modified engine installation design changes resulting from the Project Sure Fire. The tests were conducted during the period 15 January 1966 through 2 March 1966 at the Bell Aerosystems Company test laboratories.

6.3.1 Test Plan

Dynamic tests were conducted using the XRM-4 rocket engine assembly modified to incorporate the design change installation of the XLR 81-BA-13 engine components. Engine XRM-4 was modified as required to simulate the structural mass, inertia, and dynamic characteristics of a production engine. The modified rocket engine assembly was subjected to the following tests in the sequential order as listed: functional test, shock, visual inspection, functional test, vibration, visual inspection, and functional test.

The engine test assembly was vibrated along its longitudinal axis at a constant octave sweep rate for a total duration of 45 minutes. Input was at the engine mount attach points and in accordance with the following schedule.

5 to 10 cps at 1/4 inch amplitude
10 to 16 cps at 2.0 g peak acceleration
16 to 22 cps at 3.0 g peak acceleration
22 to 100 cps at 2.0 g peak acceleration
100 to 250 cps at 3.0 g peak acceleration
250 to 500 cps at 5.0 g peak acceleration

The engine test assembly was vibrated along its lateral and vertical axes at a constant octave sweep rate for a total duration of 45 minutes in each axis.

- Input at the engine mount attach points was in accordance with the following schedule.

  5 to 6.5 cps at 1/4 inch amplitude
  6.5 to 100 cps at 1.0 g peak acceleration
  100 to 250 cps at 2.0 g peak acceleration
  250 to 500 cps at 4.0 g peak acceleration

- Input at the turbine exhaust duct tab was in accordance with the following schedule.

  5 to 6.5 cps at 1/4 in amplitude
  6.5 to 60 cps at 2 g peak acceleration
  60 to 100 cps at 3.5 g peak acceleration
  100 to 250 cps at 4.0 g peak acceleration
  250 to 400 cps at 5.0 g peak acceleration

The engine test assembly was shock tested in each of the three mutually perpendicular axes according to the following schedule.

Axis	Test Level	Duration	Direction	No. of Tests
Longitudinal	25 g	6 ms	Plus and Minus	3 in each direction
Lateral	5 g	6 ms	Plus and Minus	3 in each direction
Vertical	5 g	6 ms	Plus and Minus	3 in each direction

6.3.2 Test Configuration

The XRM-4 engine consisted of a hard mockup with mass simulated non-functional parts. The functional hardware used was peculiar to the Project Sure Fire modification since all other hardware had been previously subjected to preliminary flight rating tests (PFRT) in accordance with engine specification requirements. The sinusoidal vibration test of the engine was conducted in accordance with current LMSC engine test requirements. Figures 6-3 and 6-4 show the test article in the typical setups for conducting the vibration and shock tests, respectively. Table 6-2 shows the location, orientation, and type of accelerometers used for the vibration and shock tests.

6.3.3 Test Results

A resonant frequency search of the engine test fixture disclosed resonant points with transmissibility values ranging from less than 1.0 to approximately 9.1 to 1, which could yield overtest or undertest at certain frequencies. Rework of the fixture or fabrication of a new fixture, to meet specification LMSC 6117D requirements was waived due to schedule limitations. To compensate for the fixture resonant frequencies, and possible overtest or undertest, a triaxial accelerometer was used for input control. The accelerometer used the total vector sum from each sensing element to control the shaker input.

Testing along the three axes of vibration was conducted with no structural or functional failures of the engine hardware as demonstrated by post-test functional checks and inspection. The vibration test results are listed in Table 6-3.

Post-test inspection and functional checks revealed that no structural or functional anomalies occurred as a result of the shock tests. Figure 6-5 shows typical shock photographs taken of the shock input pulses during the shock tests. Table 6-4 shows the results of the response accelerometers during each shock test.

6.3.4 Conclusions

The engine completed vibration and shock testing with no evidence of structural or functional failures to the engine hardware as a result of test levels and durations imposed. Based on these test results and on the correlation between past dynamics tests and flight history, the structural and functional integrity of the modified engine was validated and the configuration was judged to be structurally flightworthy.

Future engine vibration tests should be conducted with a test fixture natural frequency above 400 cps to eliminate engine overtesting or undertesting at certain frequencies.

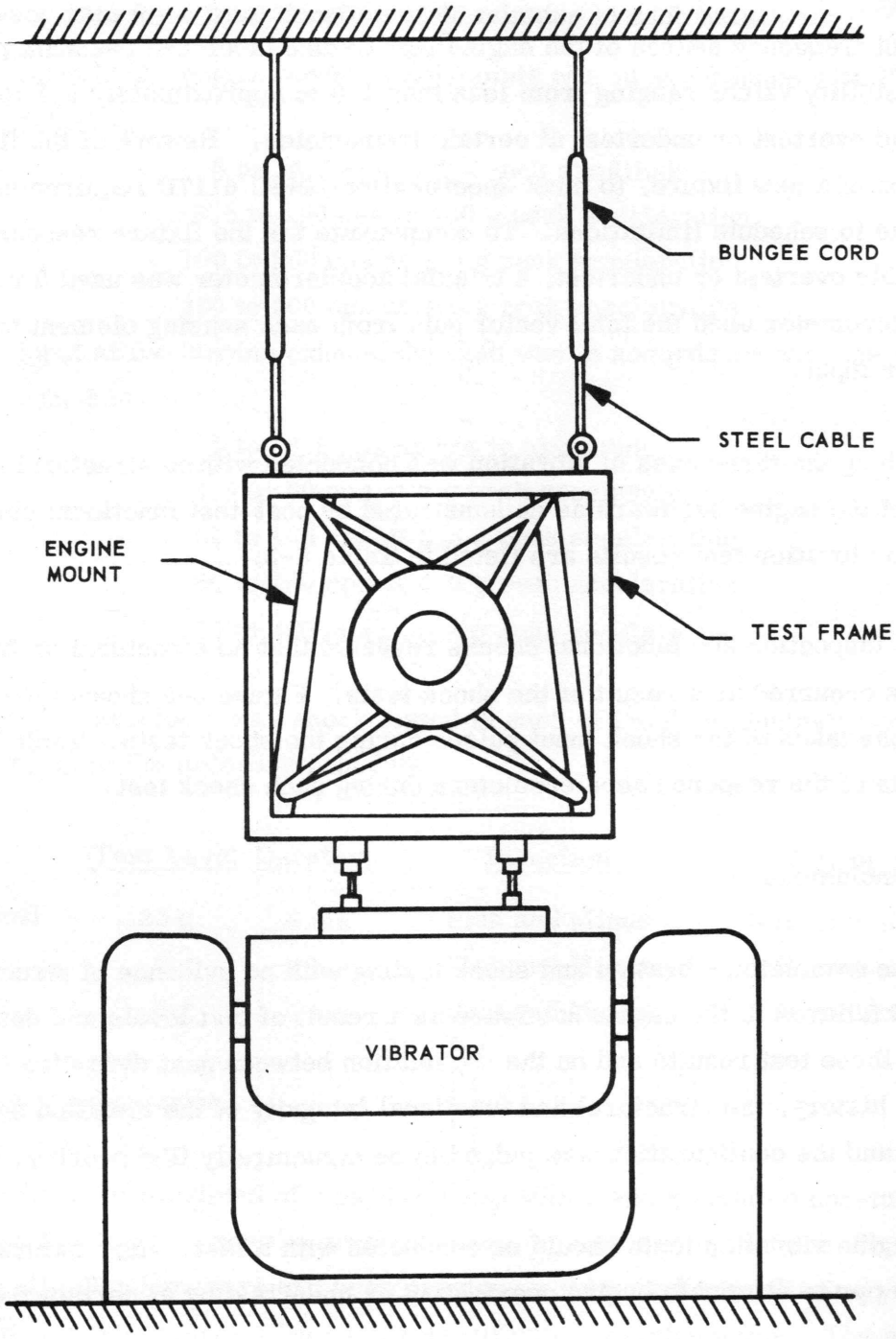


Fig. 6-3 Typical Vibration Test Setup

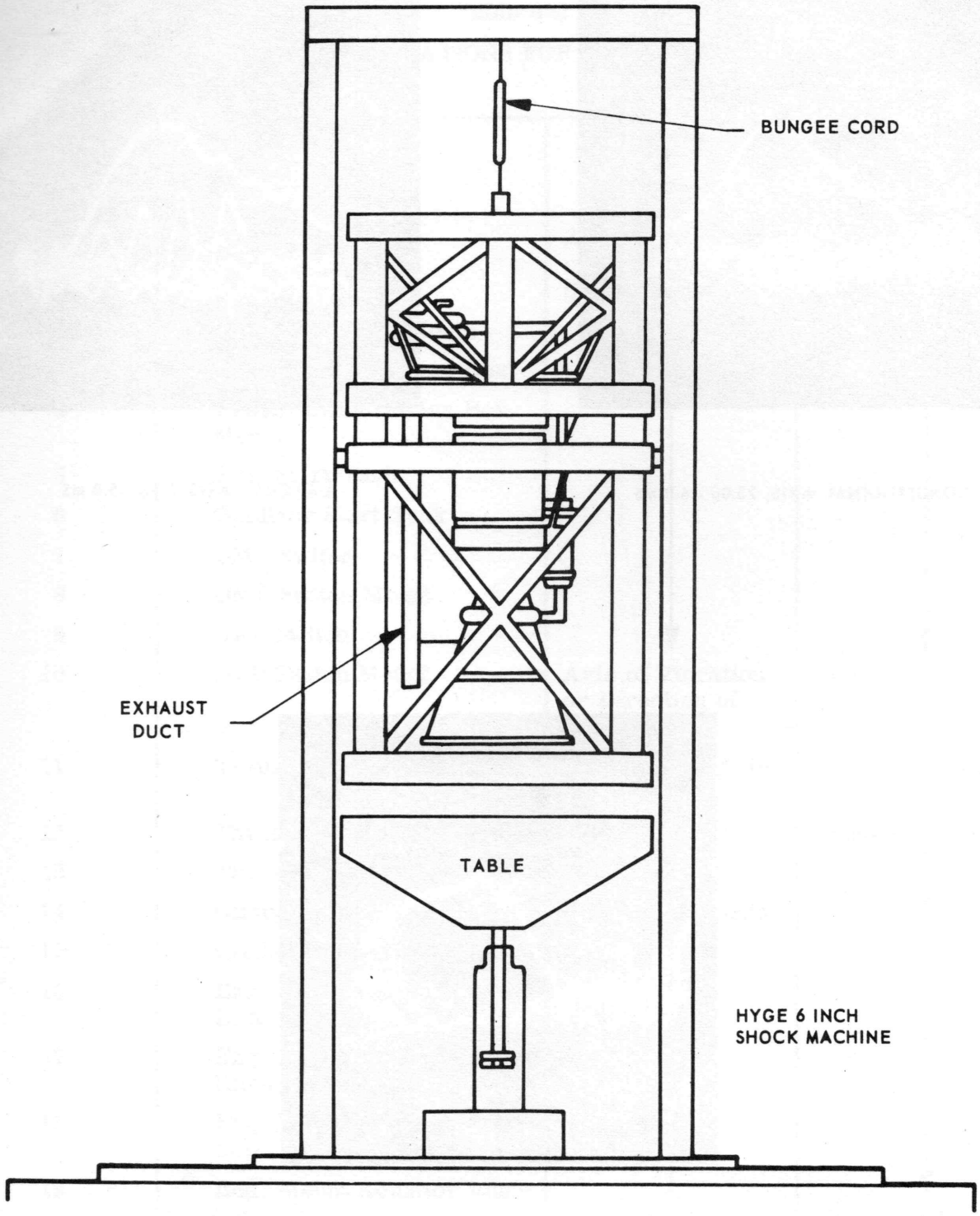
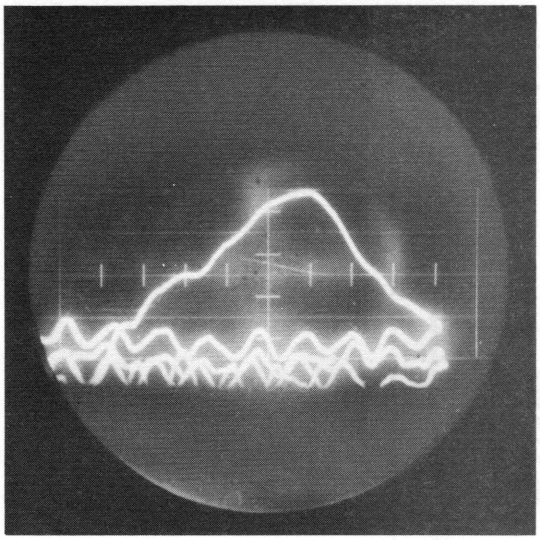
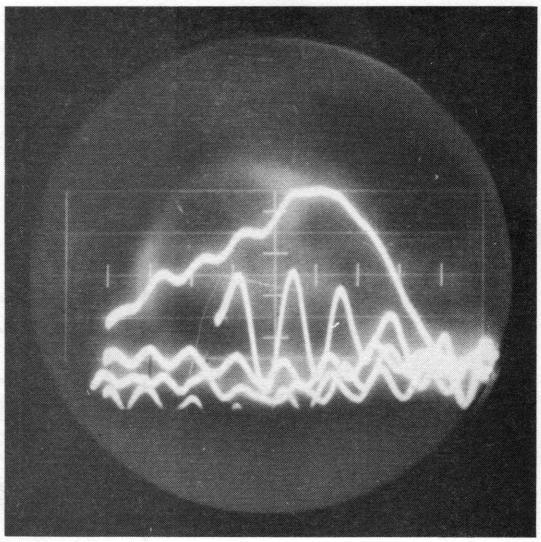


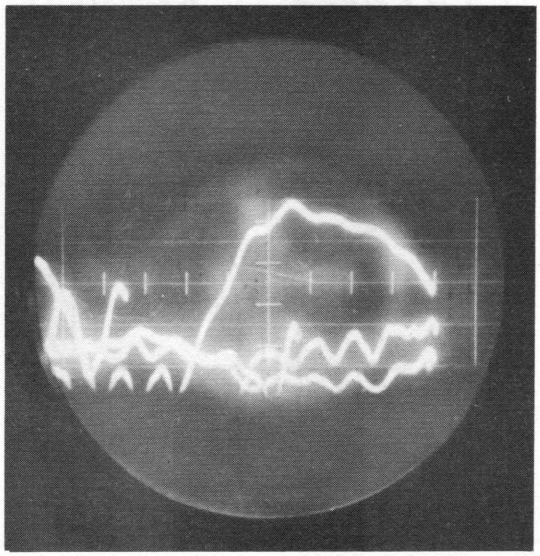
Fig. 6-4 Typical Shock Test Setup



LONGITUDINAL AXIS, 25.0g - 6.0 MS



LATERAL AXIS, 5.0g - 5.8 MS



VERTICAL AXIS, 4.7g - 6.4 MS

Fig. 6-5 Shock Test Inputs

Table 6-2

ACCELEROMETER LOCATIONS FOR VIBRATION AND SHOCK

Accelerometer Number	Location	Orientation	Type
1	Electronic Gate	Axis of Vibration or Direction of Shock	Endevco 2226
2	Electronic Gate Mount		
3	Press. Switch Relay Box		
4	Press. Switch Relay Box Mount		
5	Fuel Start Tank		
6	Oxidizer Start Tank		
7	OMP Switch		
8	OMP Switch Mount		
9	OFP Switch		
10	OFP Switch Mount	Axis of Vibration or Direction of Shock	Endevco 2226
11	Thrust Chamber Lug	Longitudinal Axis	Endevco 2242C with 2985 MI Stud
12	Thrust Chamber Lug	Vertical Axis	Endevco 2226
13	Thrust Chamber Lug	Lateral Axis	
14	Gimbal Ring – Top	Longitudinal Axis	
15	Gimbal Ring – Bottom		
16	Eng. Mount Gimbal Lug – Left		
17	Eng. Mount Gimbal Lug – Right		
18	Eng. Mount Actuator Lug – Top		
19	Eng. Mount Actuator Lug – Side		Endevco 2226
20	Thrust Chamber Lug	Longitudinal Axis	Endevco 2242C with 2985 MI Stud

Table 6-2 (Cont.)

Accelero-meter Number	Location	Orientation	Type
21	Gas Generator Injector	Axis of Injector	Endevco 2226
22	Test Fixture Near Upper Left Mount	Longitudinal Axis	
23	Test Fixture Near Upper Left Mount	Lateral Axis	
24	Gas Generator Sol. Valve Mount	Axis of Vibration (not required for shock)	
25	Fuel Dual Check Valve	Axis of Vibration (not required for shock)	
26	Oxidizer Dual Check Valve	Axis of Vibration (not required for shock)	Endevco 2226

Table 6-3

VIBRATION DATA FROM RESPONSE (OUTPUT) ACCELEROMETERS

Tests	Longitudinal Axis		Lateral Axis		Vertical Axis	
	Sine Survey	Sine Test	Sine Survey	Sine Test	Sine Survey	Sine Test
Accelerometer Numbers	g's vector					
11	3.6	11.8	4.8	12.5	1.6	3.5
12	2.4	18.6	2.7	15.5	2.8	4.6
13	2.4	15.2	2.3	5.9	5.2	5.0
14	4.1	29.1	2.4	6.8	7.0	13.5
15	–	10.6	–	10.4	–	4.0
16	2.7	10.0	1.7	6.5	10.3	5.5
17	2.3	17.8	2.4	5.8	12.4	6.2
18	1.5	9.5	7.2	25.7	14.6	20.5
19	2.0	10.0	2.7	5.9	7.5	6.3
20	2.6	23.0	2.5	11.5	5.4	2.3
21	–	24.4	–	6.8	–	9.0
22	3.0	16.2	1.3	8.0	1.9	7.1
23	0.95	7.2	1.5	6.2	4.9	5.0

Table 6-4

SHOCK DATA FROM RESPONSE (OUTPUT) ACCELEROMETERS

Tests	Longitudinal Axis						Lateral Axis						Vertical Axis					
	Plus Direction			Neg. Direction			Plus Direction			Neg. Direction			Plus Direction			Neg. Direction		
	1	2	3	1	2	3	1	2	3	1	2	3	1	2	3	1	2	3
Accelero-meter Numbers	g's Vector																	
1	41	38	38	38	30	34	8	8	7	8	8	7	8	8	8	8	8	8
2	30	34	34	30	30	27	7	7	6	6	6	5	6	7	7	6	7	7
3	35	38	38	28	28	28	8	8	7	8	8	7	7	8	8	9	8	8
4	37	37	41	37	29	29	8	8	7	8	7	7	8	8	8	9	9	8
5	37	40	40	34	31	31	7	7	6	6	6	6	7	8	7	8	8	8
6	35	35	35	35	31	35	7	7	6	7	6	6	8	8	7	8	8	8
7	38	42	42	38	35	35	9	8	8	8	7	8	7	7	7	8	7	7
8	40	36	40	33	33	33	8	7	7	7	7	7	6	7	7	5	4	4
9	33	33	33	37	37	33	8	7	7	8	8	8	7	7	7	8	7	7
10	28	32	37	32	28	37	9	9	9	10	10	10	8	9	8	10	8	10
11	45	49	49	41	41	41	–	–	–	–	–	–	5	5	3	8	8	9
12	34	34	34	26	22	26	–	–	–	5	9	7	4	5	5	7	6	8
13	28	11	11	21	14	14	–	–	–	–	–	–	–	–	–	–	–	–
14	31	31	31	8	19	66	14	13	12	10	8	9	8	9	9	8	8	8
15	6	10	10	46	40	70	8	5	3	6	5	5	4	4	3	–	–	–
16	7	9	7	38	33	19	2	2	2	2	3	3	7	8	6	7	8	7
17	–	–	–	–	–	–	–	–	–	–	–	–	5	6	6	5	6	4
18	–	–	–	–	–	–	7	6	2	5	7	7	5	4	6	4	6	6
19	–	–	–	–	–	–	7	5	5	4	7	5	3	3	3	2	2	3
20	–	–	–	–	–	–	7	8	8	14	21	22	3	3	3	3	3	3
21	–	–	–	–	–	–	7	6	5	5	7	7	2	3	3	5	5	6

6.4 ENGINE HOT FIRE TESTS

Firing tests of the Sure Fire modified engine were conducted to demonstrate flight-worthiness and to establish confidence in the operation and performance of the engine at sea level. Additional objectives of the tests were:

- To verify that preliminary pump level characteristics would be assured at the maximum expected range of Gemini mission operating pressures and temperatures.
- To certify safe engine operation at minimum and maximum voltage levels
- To verify that malfunction modes at sea level have not changed due to engine design modifications
- To evaluate shock and vibration levels at specific points on the modified engine during the engine start transient and ignition sequence.

The tests were conducted at Bell Aerosystems Company (Bell Test Center Facility), Lewiston, N. Y. during the period between 5 January and 13 February 1966. Table 6-5 shows the detailed test plan schedule of individual tests.

6.4.1 Test Configuration

Engine firing tests were conducted in test cell 2 DT, a vertical tower capable of supporting a maximum firing duration of 100 seconds (propellant tank volume limits). Temperature conditioning of the test stand could be varied between 0° F and +160° F. The engine test schedule and component configuration are shown in Tables 6-6 and 6-7, respectively. Instrumentation and data acquisition information is listed in Table 6-8. Propellant flow, functional, and electrical schematic diagrams of the modified test engine are shown in Figs. 2-1, 3-1, 6-6, and 6-7. Electrical schematic, Fig. 6-6, is applicable for test Nos. 281 through 310 and schematic, Fig. 6-7, is applicable for tests No. 311 and subsequent.

6.4.2 Modification and Acceptance Test

Engine 803 was received at BAC on 24 November 1965 in the P/N 1461969-5 configuration. The engine had been acceptance tested, had been fired on Vehicle 5001 by LMSC at Santa Cruz Test Base, and had gone through a "wet" countdown at the eastern test range launch complex. Modification consisted of replacing the XLR 81-BA-13 main propellant valves with YLR 81-BA-11 valves and addition of an oxidizer manifold pressure switch (OMPS) and an oxidizer feed pressure switch (OFPS) to assure an oxidizer pre-flow start. These modifications required addition of a pressure switch relay box and an electrical harness change to accommodate the addition of the pressure switches to the electrical system.

During pre-fire inspection, contamination was found which subsequently led to a thrust chamber flush. At this time it was found that the small (0.026 in. dia.) fuel barrier holes were not impinging properly. The results of the reacceptance test indicated that the chamber would be acceptable for FWD testing. The engine was reacceptance tested on 23 December 1965 in the BAC P/N 8247-475200-11 configuration. (Note: This included the BAC P/N 8247-472283-5 electronic gate and the BAC P/N 8247-475491-1 cable assembly.) A design change was made that resulted in the electronic gate receiving its power supply through the vehicle aft safe arm J -box (ASA J-box). This created a new cable assembly (BAC P/N 8247-475481-1), which was installed for engine post-fire functional checks.

6.4.3 Installation and Pre-Fire Servicing

During the engine functional checks, fuel gas generator solenoid valve (GGSV) S/N 111 exhibited an excessive N_2 leakage of 9.35 cc/hr. at 1500 psig nitrogen pressure (maximum allowable is 6.0 cc/hr). Valve S/N 111 was replaced with S/N 115 prior to the calibration test series.

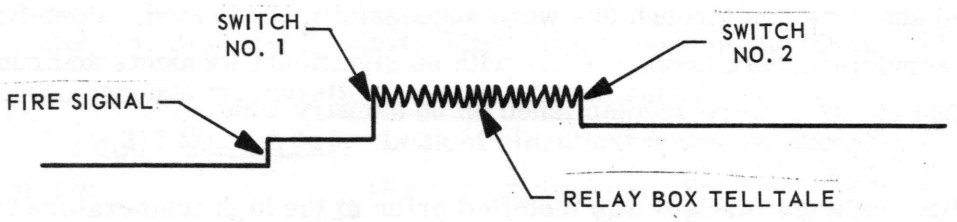
The engine was installed on the test cell tower and during the test cell/engine electrical compatibility checkouts, a short circuit was inadvertently made across pins A and B of the P-6002 connector. The short circuit was caused by a faulty pin on the test cable being used for checkout. Since possible damage could have occurred to the

pressure switch relay box when the short occurred, both the cable assembly S/N 3 and the relay box S/N 3 were replaced by cable assembly S/N 6 and relay box S/N 10. Significant modifications affecting the engine interface to facilitate test instrumentation included:

- Removal of the OMPS sensing line and installation of a test line manifold and transducer
- Installation of a test harness shunt packs between connectors P-6023 and J-6023 (fuel GGSV) and between connectors P-6024 and J-6024 (oxidizer GGSV).

6.4.4 Fixed Thrust Calibration Tests

This first series of tests was intended to verify engine sequencing and operation and to calculate engine power level and sea level performance. On 16 January 1966 a 90-second firing (run no. 281) was accomplished followed by a malfunction on the next start (run no. 282). Review of the data indicated no power had reached the engine. Subsequent trouble shooting isolated the problem to an electrical short in the facility-simulated ASA J-box. The simulated ASA J-box was replaced and an improved mounting installation was incorporated. Post-fire and pre-fire servicing was conducted and on 20 January 1966 runs 283 through 296 were accomplished. These tests consisted of a 30-second firing and 13 firings of 2 seconds duration each, with varying coast times. Review of the test data indicated that it was difficult to determine if the oxidizer manifold pressure switch (OMPS) or oxidizer feed pressure switch (OFPS) commanded the pilot operated solenoid valve (POSV), due to chatter in the OFPS and the slow response of the pressure traces. The characteristic observed is shown as follows:



For the next firing the OFPS was removed from the command circuit and a switch PIP monitoring circuit installed. Run No. 297, a 2-second bit impulse run was performed and results demonstrated that the OMPS actuated at approximately 0.150 second before the OFPS. Prior to the next run, the OFPS was connected back into the command circuit and both switches were wired to give a switch PIP indication. Also, the OMPS sensing line was changed to flight configuration (1/8 inch tubing, approximately 33 inches long with one transducer, 0-50 PSIG).

On 22 January 1966 a 90-second firing was conducted (run No. 298) which completed the fixed thrust calibration tests.

6.4.5 High and Low Temperature Tests

High and low temperature tests were conducted to verify proper operation of the modified engine over the specification temperature range. While conducting the pre-fire engine servicing, a gas leakage was noted at the oxidizer pump discharge outlet flange and the seal was replaced. On 26 January 1966, temperature conditioning was initiated, starting the 24-hour soak at +140° F. The high temperature soak was aborted after 10 hours due to low oxidizer start tank pressure. Investigation revealed that an error was made in charging the start tank which resulted in a gas leak and an actual pressure of 770 psig existed in lieu of the required pressure of approximately 1000 psig.

After correcting the pressure discrepancy, difficulty was experienced with the steam circulation equipment (ambient weather was -10° F) and it was agreed to conduct the cold temperature tests first. On 29 January 1966, the 24-hour soak at +10° F was completed and runs 299 through 301 were successfully performed. Post-fire and pre-fire servicings were accomplished with no significant incidents and runs 302 through 304 at +32° F were accomplished on 30 January 1966.

The facility electrical system was modified prior to the high temperature test to provide post shutdown monitor capability of the OMPS and OFPS deactuation sequence.

The remaining 14 hours of the 24-hour soak period were scheduled and on 4 February 1966 runs 305 through 307 at +140° F were performed. The following day, the +90° F firings, runs 308 through 310 were accomplished.

6.4.6 Safety Limits Tests

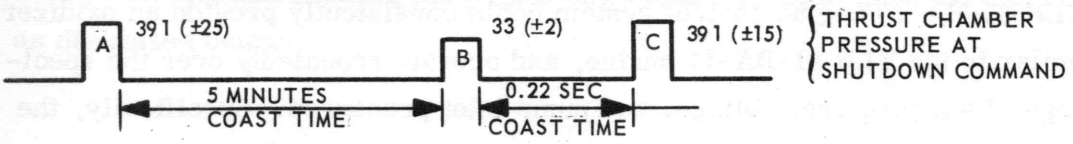
The purpose of the safety limits tests were to verify that the modified engine would operate properly over the range of specification voltages. The OFPS S/N A-10573 was replaced with S/N A-10571 because the 1/8 inch threads were stripped. The LMSC flight pressure transducer (for fuel valve actuation pressure) was installed, and relay box S/N 10 was replaced with pressure switch junction box S/N 001.

On 9 February 1966, four safety limit tests, runs 311 through 314 were accomplished. During run No. 312, the Beckman tape jammed after eight seconds, resulting in a loss of data. Also lost on the oscillograph was oxidizer suction pressure, oxidizer flow rate, and fuel valve actuation pressure.

6.4.7 Malfunction Tests

The first malfunction test (fuel valve fails to open on start, run No. 315), was accomplished in series with the safety limit tests. The run was terminated after 3 seconds by timer to reduce the amount of oxidizer flow in the test stand area. The speed trace indicated that an overspeed shutdown would have occurred within 0.5 seconds of additional engine operating time. The remaining eight malfunction tests were grouped into two test series to minimize engine servicing requirements between runs and reduce the test schedule.

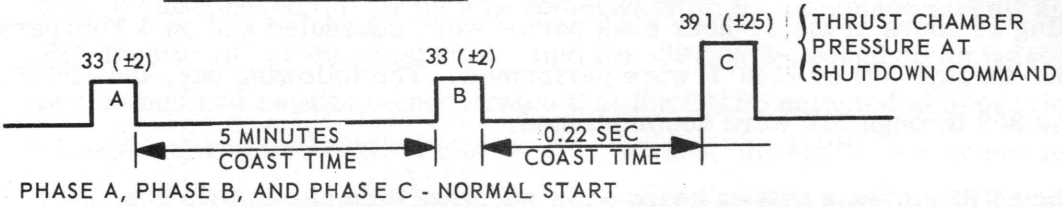
- Run No. 316. OFPS start; POSV fails to open on shutdown. Phase A only. 20 seconds run duration. Accomplished satisfactorily.
- Runs 317 through 319. Loss of electrical power on Start No. 1



PHASE A - NORMAL START

PHASE B AND PHASE C - POSV ENERGIZED ON START (FUEL LEAD)

- Runs 320 through 323. Loss of power on Start No. 2.



PHASE A, PHASE B, AND PHASE C - NORMAL START

- Run No. 324. Simultaneous propellant exhaustion. Phase A only – 20 seconds duration.

The OMPS was removed from the command circuit and the flight OFPS sensing line was installed during run no. 316. At approximately 0.22 seconds following the shutdown of run no. 320 (Phase A of loss of power on start no. 2), the engine was inadvertently fired (run no. 321), with shutdown initiated by loss of power at a chamber pressure of 397 psig. This produced one extra run, and was later traced to a malfunction of the malfunction test panel.

6.4.8 Post Fire Servicing

On 13 February 1966, direction was received to terminate the sea level FWD Program and ready the engine for shipment to AEDC. The engine was flushed, final functional checks were conducted, and discrepancies found were resolved (refer to Failure Analysis & Corrective Action Summary in Section 11).

Since the engine was committed to the AEDC Program, the final disassembly and inspection of the following components was deferred until return of the engine from AEDC: oxidizer manifold pressure switch, oxidizer feed pressure switch, fuel valve, and oxidizer valve.

6.4.9 Engine Performance and Operation Test Results

The engine sea level hot fire tests performed on engine 803 demonstrated that the modified XLR 81-BA-13 engine control system could consistently provide an oxidizer preflow similar to the YLR 81-BA-11 engine, and operate repeatedly over the specification range of temperature, voltage, and pump inlet pressures. Specifically, the addition of the oxidizer manifold pressure switch (OMPS), and substitution of the YLR-81-BA-11 main oxidizer and fuel valves resulted in an average sea level

oxidizer preflow of 4.7 pounds. The average sea level oxidizer preflow of the YLR 81-BA-11 engine is 4.8 pounds. A summary of the engine sea level FWD tests is presented in Table 6-9.

An evaluation of the engine transient data indicates excellent engine operation within predicted limits throughout the range of environments to which the engine was subjected. Two trends of significance noted in correlating the data are as follows:

- The engine consistently exhibited an oxidizer postflow of approximately 3 pounds less (28 pounds vs. 31 pounds) than the YLR 81-BA-11 engine. This difference is attributed to the faster closing time of the gas generator oxidizer valve (GGOV) on the XLR 81-BA-13 engine relative to the gas generator bi-propellant valve on the YLR 81-BA-11 engine. When the engine main fuel valve closes, the engine speed increases until the turbine manifold pressure decays as a result of one or both of the propellants being terminated to the gas generator. Then oxidizer postflow becomes a function of the peak turbine speed achieved since oxidizer flow is proportional to turbine speed and the turbine speed decay rate is repeatable. The average closing time of the XLR 81-BA-13 GGOV, which closes first, is 0.110 seconds, compared to approximately 0.150 seconds for the YLR 81-BA-11 bi-propellant valve.
- At both the 19.5 and 22.5 volt power input levels to the engine, a trend was detectable in the gas generator solenoid valve opening times, although the effect on engine start transient sequencing is not significant. Three 19.5-volt tests at ±140° F and one 19.5-volt and one 22.5-volt test at 65° F indicated an average GGOV and GGFV opening time delay of 0.015 and 0.050 seconds respectively, compared to the other tests at 26 volts and above.

The malfunction tests demonstrated that the engine control system was capable of safely starting and shutting the engine down under various imposed malfunction modes, as discussed below:

- Engine Main Fuel Valve Fails to Open – This test indicated that the electronic gate would safely shut the engine down, due to turbine overspeed, in 2 to 3 seconds after the time at which the fuel valve should open.
- Oxidizer Feed Pressure Switch Start – In this test, simulating an OMPS which fails to actuate, the OFPS provided a 17 pound oxidizer preflow and the engine started without detrimental engine operation. Reliable ignition with a larger oxidizer lead was also demonstrated during AEDC tests (Refer to Section 7). Turbine speed rose to 23,750 rpm during the start transient which is approximately 2000 rpm higher than a normal OMPS start.
- Power Interruption Series – The engine restarted and shut down safely without any apparent detrimental effects on the engine hardware.
- POSV Fails to Open on Shut Down – The main fuel valve remained open at shutdown signal while the gas generator valve closed and terminated combustion in the gas generator. This resulted in a more rapid than normal turbine speed decay rate at shutdown due to the additional loading on the pumps caused by the fuel flow to the thrust chamber. At a turbine speed of 7500 rpm and a thrust chamber pressure of 90 psia, the fuel valve closed due to the decay in fuel pump discharge pressure, and the oxidizer valve closed at 6500 rpm due to the decay in oxidizer pump discharge pressure. There were no deleterious effects observed during this abnormal shutdown transient.
- Simultaneous Propellant Depletion – The electronic gate turbine overspeed circuit shut the engine down safely, limiting the maximum turbine speed to a value of 35,300 rpm.

6.4.10 Engine Performance and Operation Test Conclusions

The modified XLR 81-BA-13 engine start, steady state, and shutdown characteristics were all satisfactory and within predicted limits during the sea flightworthiness tests. demonstrating that the modified design was ready to be tested at simulated altitude conditions.

All procedural changes for the modified engine were checked out and verified, and declared usable for the simulated altitude tests and for flight engine checkout at LMSC, Sunnyvale, and Eastern Test Range.

6.4.11 Dynamic Data Test Results

The engine and facility thrust mount were instrumented with dynamic measurement transducers as shown in Table 6-10 and Fig. 6-8. Engine vibration, force, and pressure measurements were monitored during all sea level FWD hot fire tests.

Test dynamic data results are presented in Tables 6-11 and 6-12. Table 6-11 is a summary of the dynamic data at thrust chamber ignition on engine hot fire tests 281 and 283 through 324. Accelerometer data in Table 6-11 are the peak g's recorded at thrust chamber ignition. Average peak g's for each run are listed to give a comparison between runs for different hardware and propellant temperature conditions. Using accelerometer A-1 (mounted on the thrust chamber lug) as a reference accelerometer, the minimum and maximum peak g's from the 43 engine firings were:

- Minimum – 16 g's
- Maximum – 108 g's

The average peak g's for each run varied from 41 to 79. The high peak g average resulted from the malfunction test 319 which was the second pulse of a multiple power interruption test with 0.22 seconds between the first shutdown signal and restart command. The start transient total force levels from the load cell data varied from 9409 to 14,105 pounds. The load cell data from test 281 was not considered valid due to calibration discrepancies. The mean total force from the available data was 11,757 pounds, which is near predicted for sea level conditions.

Table 6-12 shows the results of the Power Spectral Density (PSD) data from 9 of the 43 tests that represent different propellant inlet temperature conditions from 0 to +143° F. The data shows a spread of 0.5 to 17 g (RMS) for the engine test firing for a 0.5 second period beginning shortly after thrust chamber ignition.

The Power Spectral Density data shown in Table 6-12 and Figs. 6-9, 6-10 and 6-11 are typical results of the sea level engine FWD tests and indicate average levels significantly less than the Specification LMSC 6117D test requirements of 18.7 g (RMS). These PSD plots show distinct peak frequencies occurring around 850, 1500, and 1700 cps. Most of the energy represented by the area under each PSD plot occurs at these predominant frequencies which, as expected, correlate with the engine pump impeller frequencies.

An evaluation of the dynamic test results does not indicate a trend of thrust chamber ignition shock as a function of temperature at sea level conditions. However, the high response chamber pressure measurement indicated a more gradual initial rise rate (0.070 seconds from 0 to 70 psig) at higher temperatures. Test 318 was the start for a power interruption fuel lead start sequence and no indication of an ignition delay and no significant difference in ignition accelerations or thrust loads were observed, however, the highest chamber pressure rise rate (0.035 seconds from 0 to 70 psig)was observed on this test. The second pulse of the fuel lead power interruption (test 319) indicated the highest average ignition peak shock levels. However the chamber pressure characteristics, which were affected by the short coast periods wherein residual fuel is retained and reacts with the oxidizer post flow, do not permit a valid correlation of chamber pressure rise rate for this test.

6.4.12 Dynamic Data Test Conclusions

- The sea level engine FWD dynamic test results, throughout the range of test conditions, indicate no adverse loading or accelerations during the engine start transient or operation. Therefore, within measurement and analytical capability, adverse conditions during the fuel lead start were not observed at sea level.
- The general review of steady state engine operation vibration levels during these FWD tests indicate levels approximately equal to the specification LMSC 6117D test requirements.

- The average thrust chamber ignition peak g's (53 g's) recorded with the engine hard mounted installation were greater than the shock test levels of 40 g's specified in LMSC 6117D, environmental test specification for the vehicle component installations. An extensive analysis of each ignition accelerometer transient would be required to present the total energy levels at ignition for each accelerometer transient. However, an evaluation of the high peak accelerometer data indicates the ignition shock total energy levels obtained with the engine FWDT hard mounted installation were lower than the LMSC 6117D specification test requirements of 1/2 sine wave, 40 g's peak for 8 millisecond duration.
- The engine FWD test ignition average peak acceleration levels are greater than the corresponding average longitudinal acceleration peak levels recorded during the XLR 81-BA-13 and YLR 81-BA-11 engine acceptance tests (Ref. paragraph 5.6). However, the differences in facility thrust structure, engine attitude, accelerometer installation techniques, and rigidity of gimbal actuators compared with solid links prohibit a direct correlation of the accelerometer test results even at similar test conditions.
- The dynamic test results indicate that test conditions per se did not impose significant effects on engine dynamic operation, over the range of the sea level FWD conditions tested.

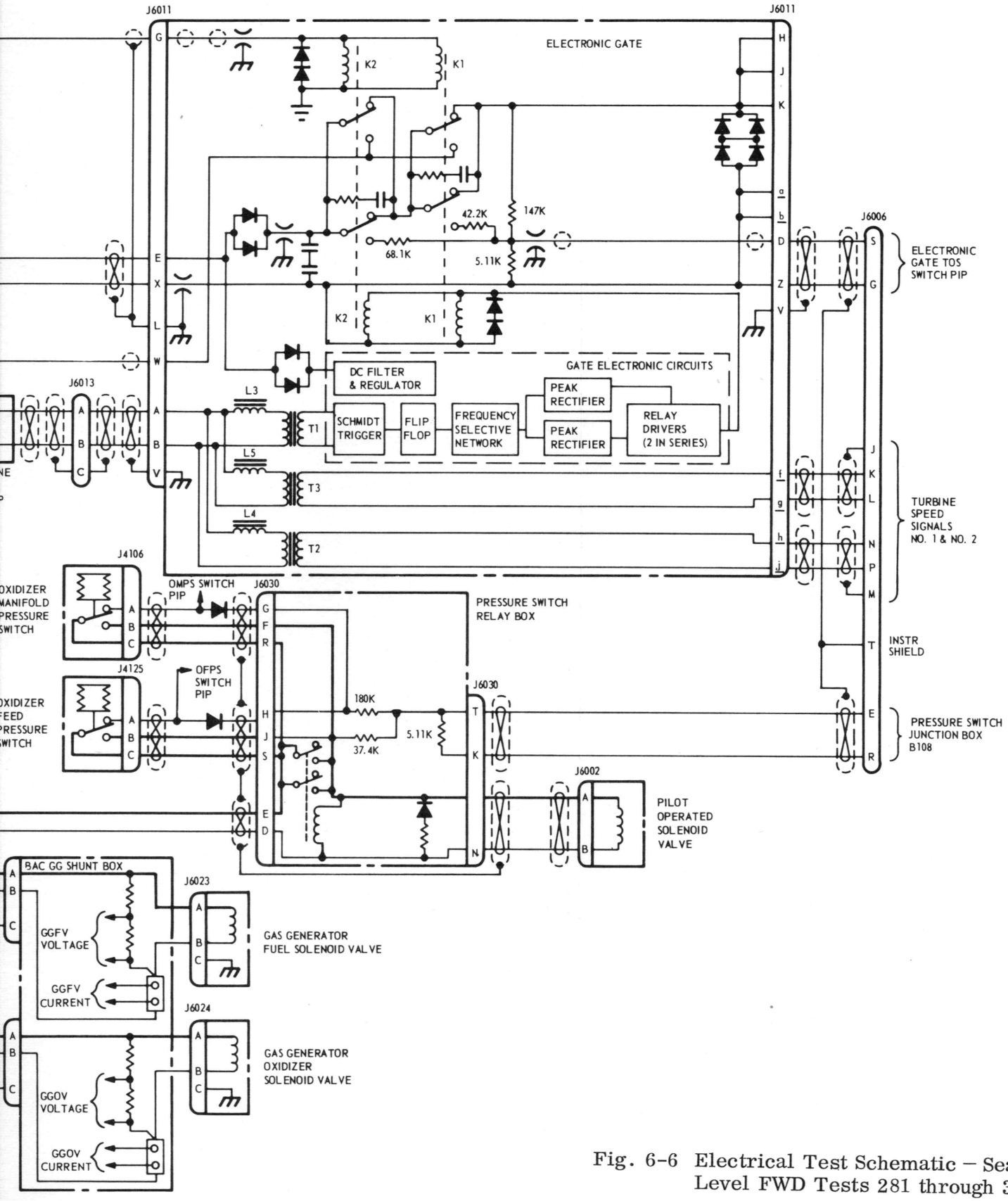
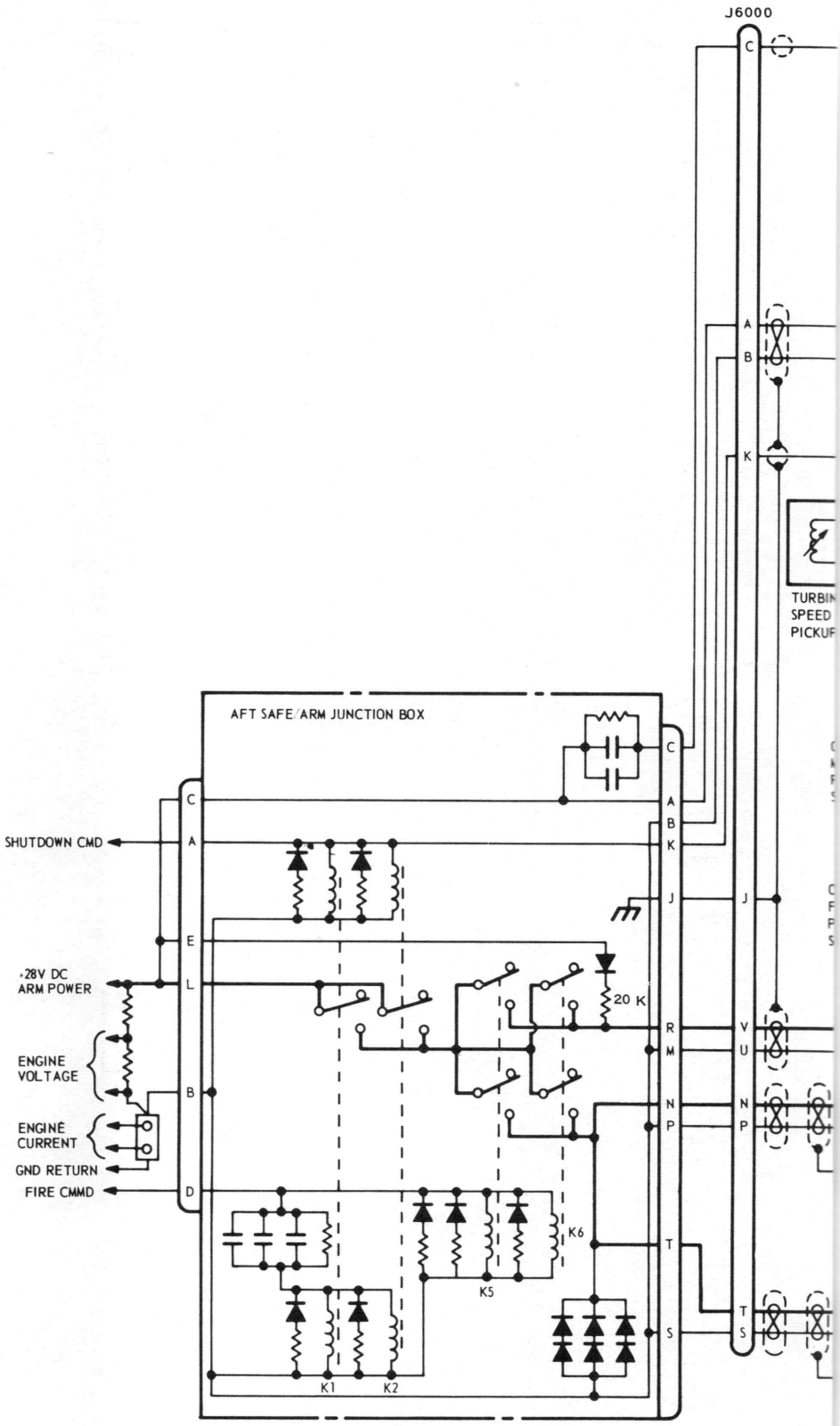


Fig. 6-6 Electrical Test Schematic – Sea Level FWD Tests 281 through 310


J6000
AFT SAFE/ARM JUNCTION BOX
SHUTDOWN CMD
+28V DC
ARM POWER
ENGINE
VOLTAGE
ENGINE
CURRENT
GND RETURN
FIRE CMMD
20 K
K1
K2
K5
K6
TURBIN
SPEED
PICKUP

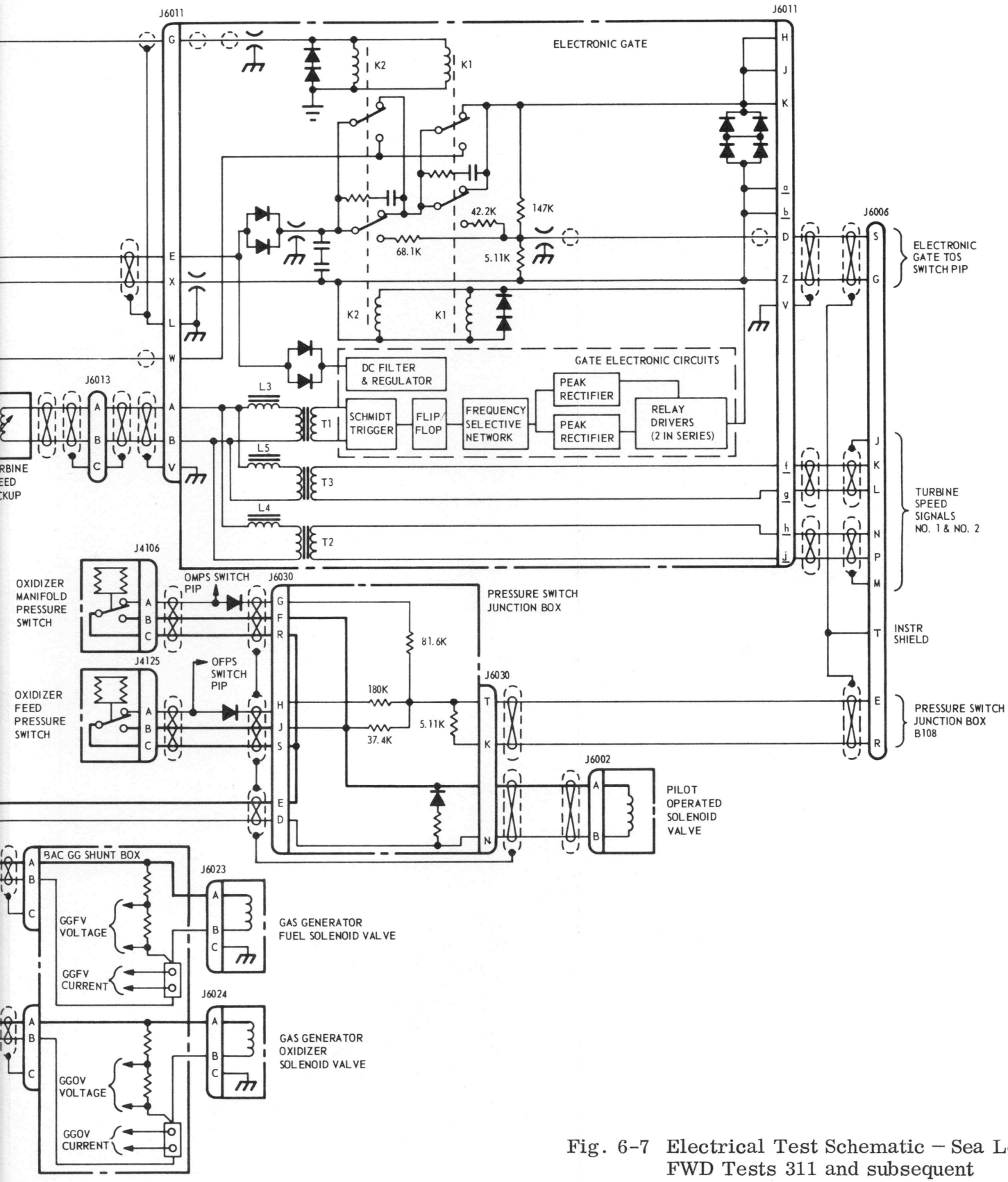
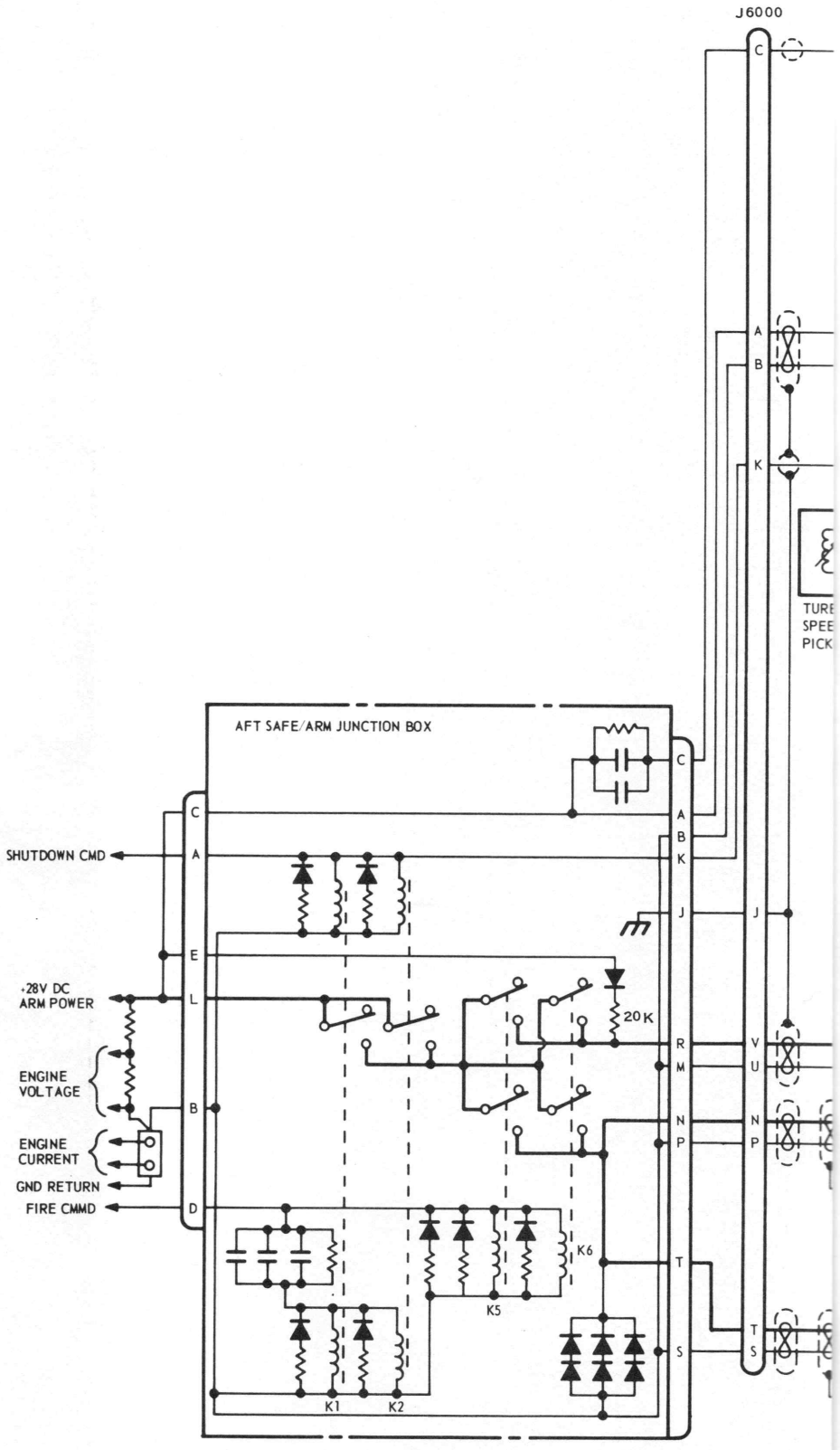


Fig. 6-7 Electrical Test Schematic – Sea Level FWD Tests 311 and subsequent


J6000
AFT SAFE/ARM JUNCTION BOX
SHUTDOWN CMD
+28V DC
ARM POWER
ENGINE
VOLTAGE
ENGINE
CURRENT
GND RETURN
FIRE CMMD
20K
K1
K2
K5
K6
C
A
B
K
J
E
L
D
R
M
N
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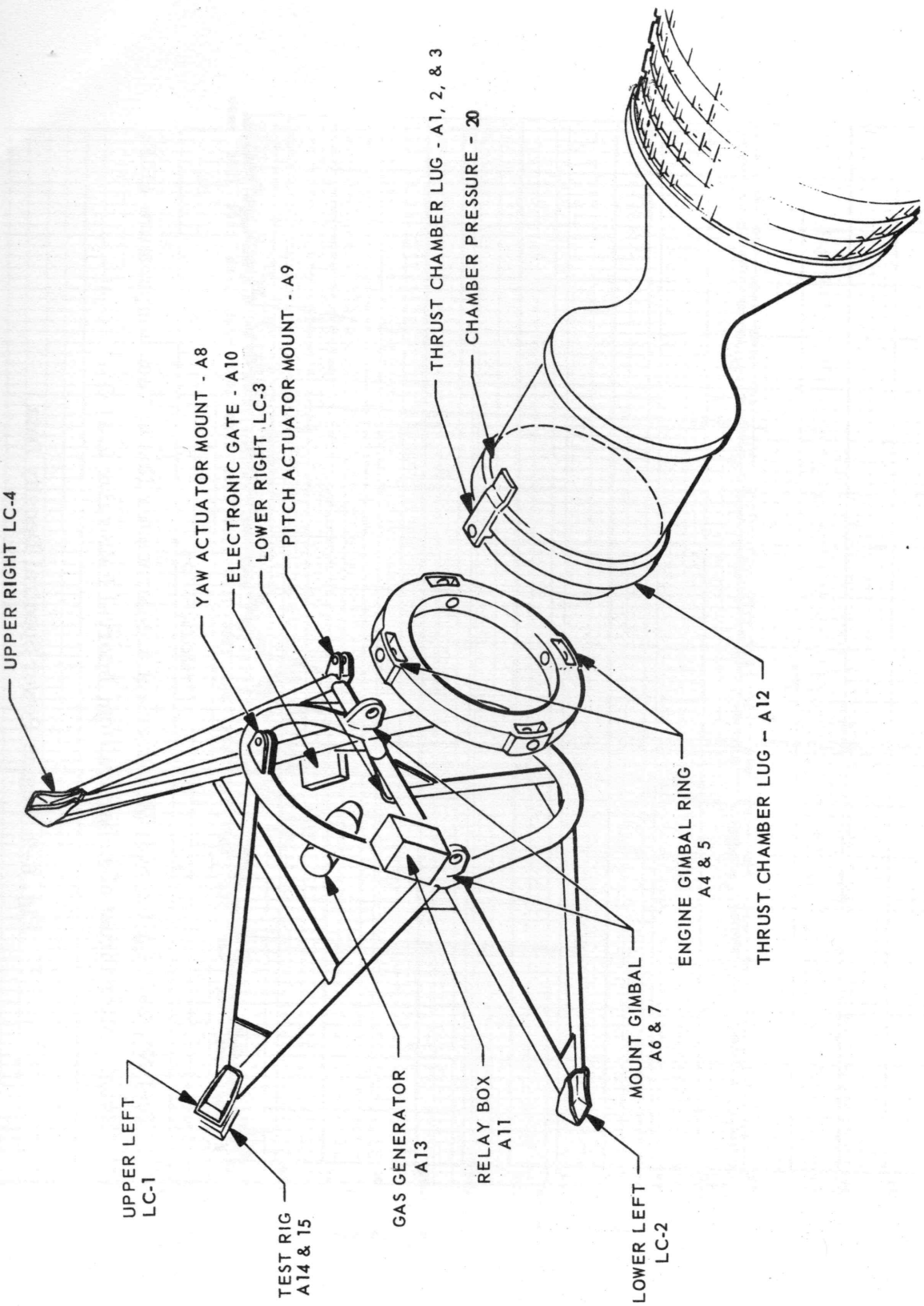
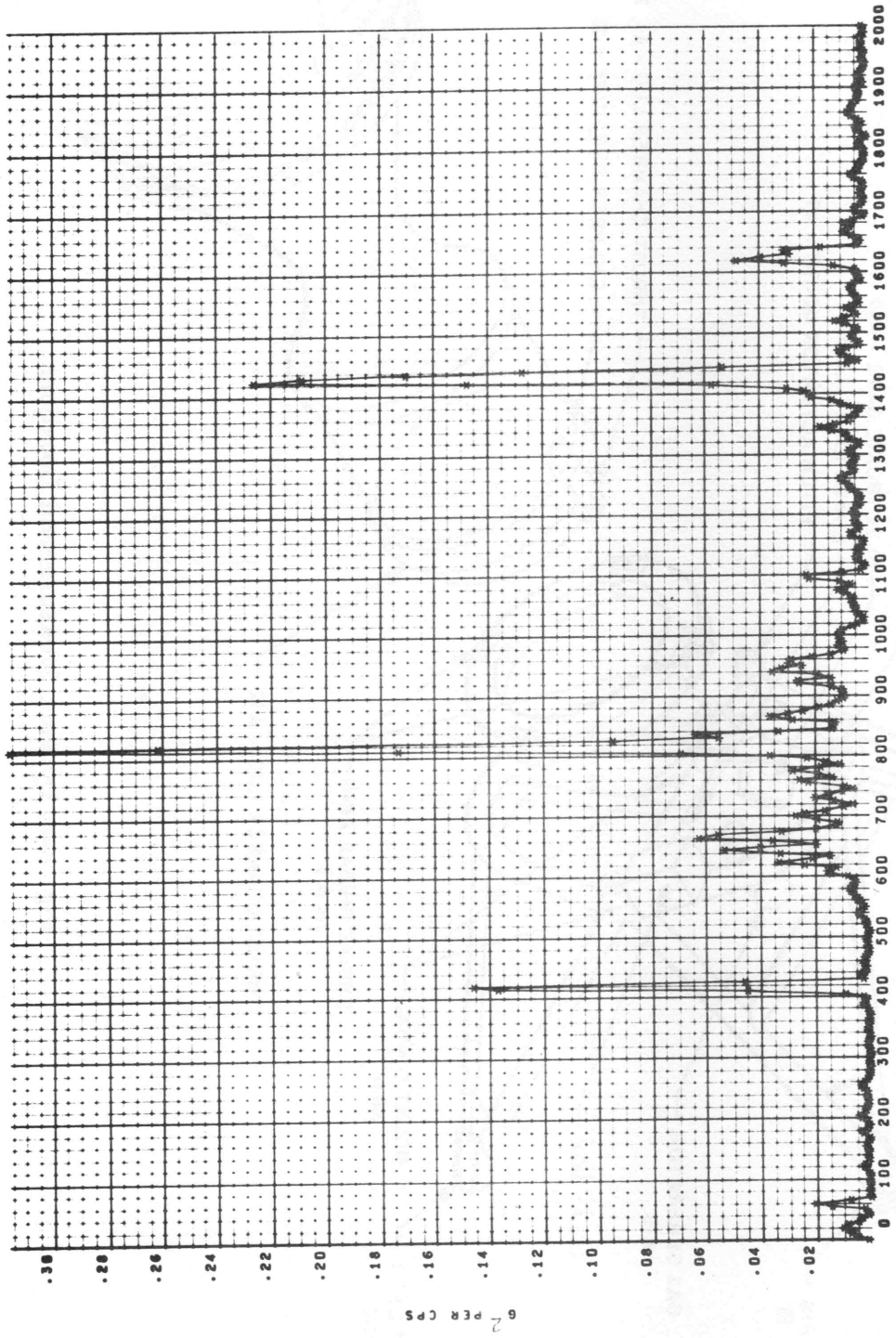
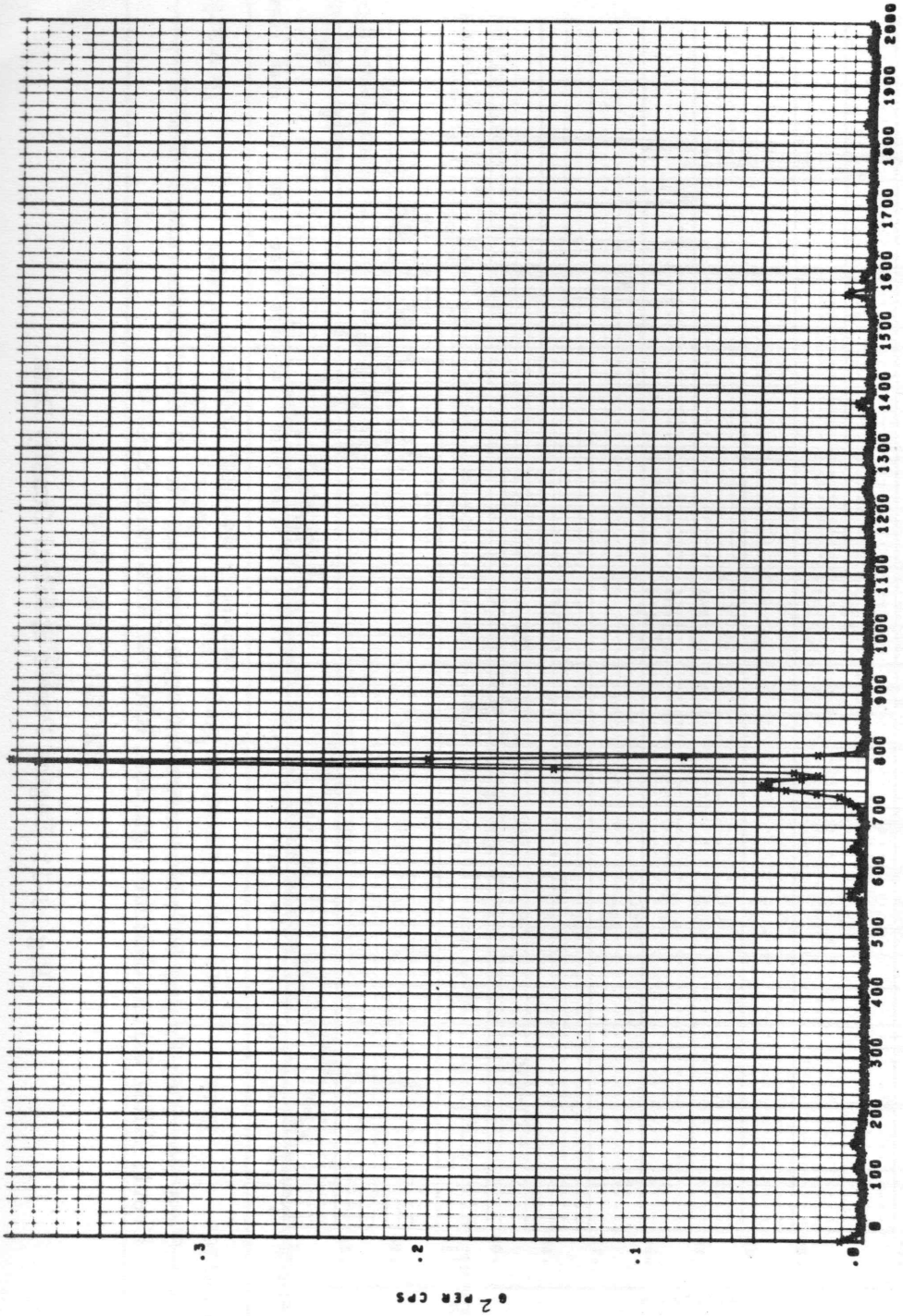

UPPER RIGHT LC-4
YAW ACTUATOR MOUNT - A8
ELECTRONIC GATE - A10
LOWER RIGHT LC-3
PITCH ACTUATOR MOUNT - A9
THRUST CHAMBER LUG - A1, 2, & 3
CHAMBER PRESSURE - 20
UPPER LEFT LC-1
TEST RIG A14 & 15
GAS GENERATOR A13
RELAY BOX A11
LOWER LEFT LC-2
MOUNT GIMBAL A6 & 7
ENGINE GIMBAL RING A4 & 5
THRUST CHAMBER LUG - A12

Fig. 6-8 Dynamic Measurement Locations



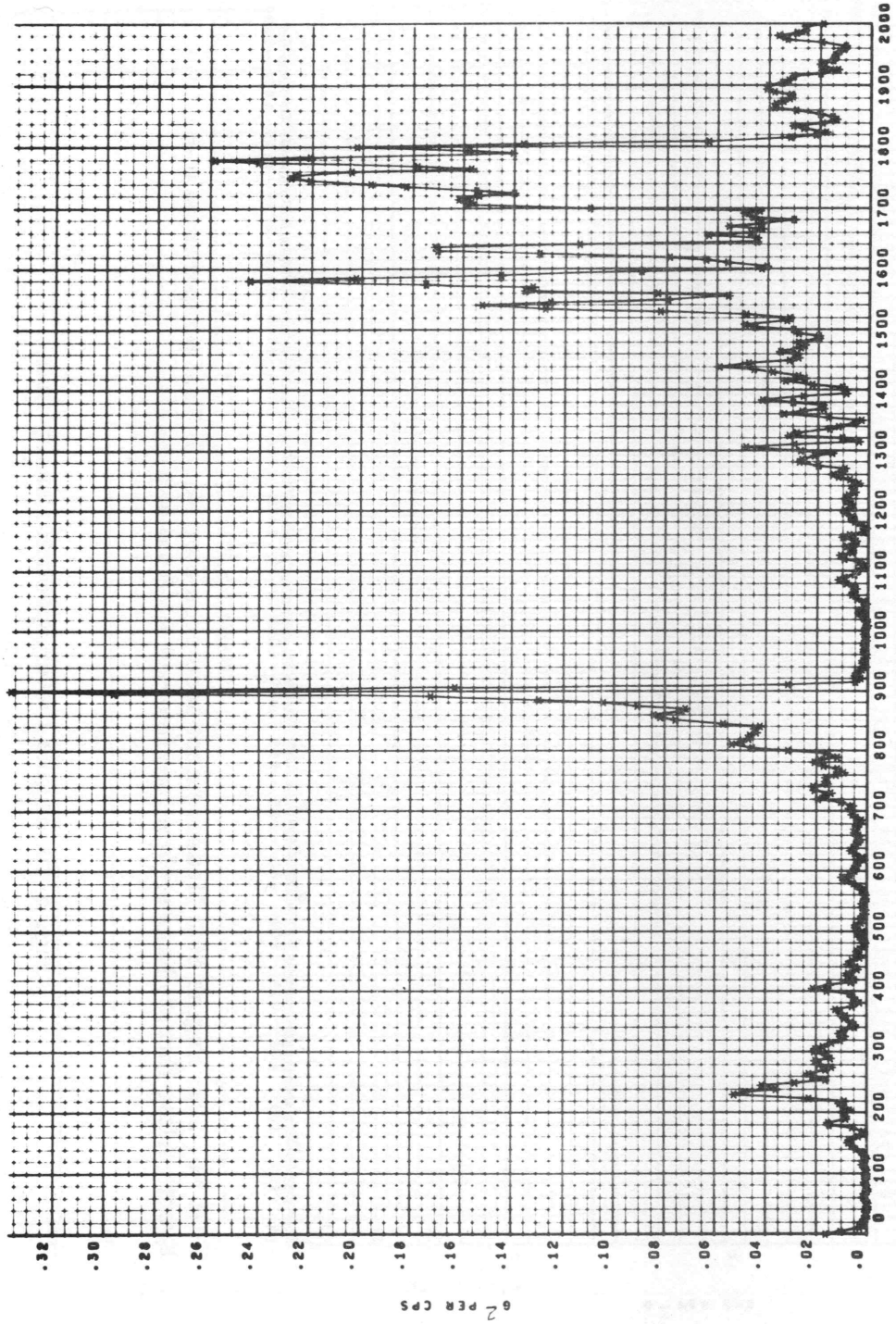
Test No. 297; Engine 803; Measurement accelerometer No. 6; Test temperature 41°F;
Event: After thrust chamber ignition; Interval RMS value 5.31 G

Fig. 6-9 Engine Power Spectral Density Data



Test No. 299; Engine 803; Measurement accelerometer No. 3; Test temperature 0° F;
Event: After thrust chamber ignition; Interval RMS value 3.25 G

Fig. 6-10 Engine Power Spectral Density Data



Test No. 306; Engine 803; Measurement accelerometer No. 9; Test temperature +143° F;
Event: After thrust chamber ignition; Interval RMS value 8.43 G

Fig. 6-11 Engine Power Spectral Density Data

Table 6-5

ENGINE TEST PLAN

Temperature								
Test Series	Date	Run No.	Duration	Prop	Hdwe	Steady State Suction Press	Voltage	Remarks
Acceptance	12-23-65	H2-435A	60 sec	60	60	24 (±5) psia	24 (±1)	
		B	2					Hardware temperature function of previous firing
		C	2					
		D	2					
		E	60					
Calibration	1-16-66	2DT0281	90	60	60	24 (±5) psia	24 (±1)	ASA J-box replaced prior to run No. 283
		282	Malf.		N.A.			
	1-20-66	283	30		60			
		284	2		N.A.			
		285						
		286						
		287			N.A.			
		288			<100			
		289			<100			
		290			N.A.			
		291			N.A.			
		292			N.A.			
		293			<100			
		294			<100			
		295			N.A.			
		296			N.A.			
	1-21-66	297	2		60			OFPS out of command circuitry for run No. 297 OMPS flight sensing configuration installed
	1-22-66	298	90		60			
Low Temp.	1-29-66	2DT-299	30	10	10	24 (±5) psia	19.5 (±0.5)	
		300	2	10	N.A.			
		301	2	10	N.A.			

Table 6-5 (Cont.)

Temperature								
Test Series	Date	Run No.	Duration	Prop	Hdwe	Steady State Suction Press	Voltage	Remarks
Low Temp.	1-30-66	20T-302	30	32	32	24 (±5) psia	30.5 (±0.5)	
		303	2	32	N.A.			
		304	2	32	N.A.		30.5 (±0.5)	
High Temp.	2-4-66	305	15	140	140		19.5 (±0.5)	
		306	2	140	N.A.			
		307	2	140	N.A.		19.5 (±0.5)	
	2-5-66	308	30	90	90		26 (±1)	
		309	2	90	N.A.			
		310	2	90	N.A.		26 (±1)	
						Fuel Ox		
Safety Limits	2-8-66	311	20	60	60	24 8	28 (±1.0)	Installed POSV J-box prior to run 311
		312	20		N.A.	11 8	19.5 (±0.5)	
		313	20		N.A.	50 32	30.5 (±0.5)	
		314	20		N.A.	11 24	22 (±1.0)	
Malfunction	2-8-66	315	3		N.A.	24 (±5) psia	24 (±1)	Fuel valve fails to open.
	2-12-66	316	20		60			OFPS start – fuel valve fails to close on shut-down
		317	–		N.A.			Loss of Power On Start I
		318	–					
		319	–					
		320	–					Loss of Power On Start II
		321	–					
		322	–					
		323	–					
		324	20					Simultaneous Exhaust

Table 6-6

ENGINE TEST SCHEDULE

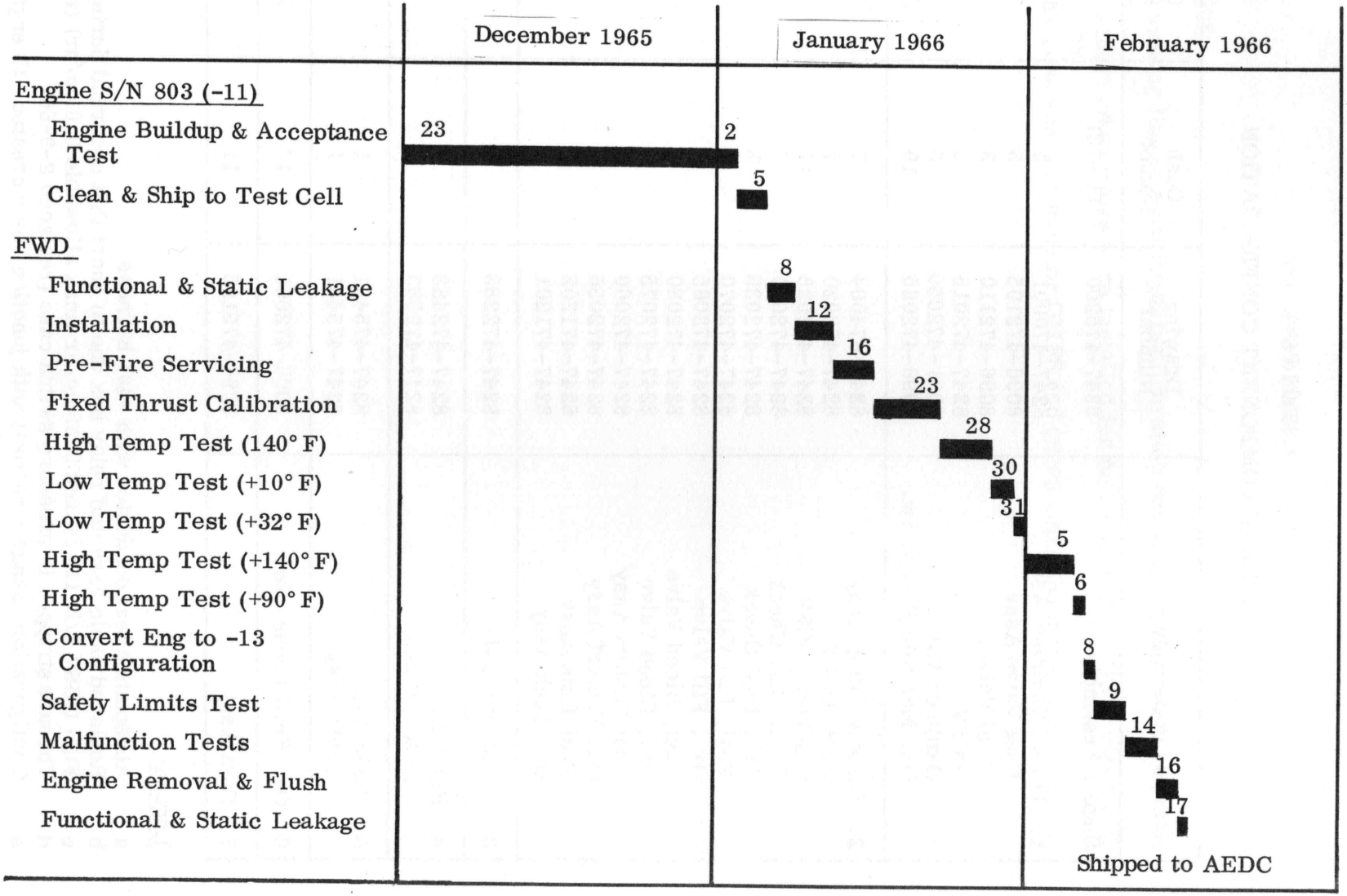


Table 6-7

ENGINE COMPONENT CONFIGURATION

Assembly	Drawing Number	Dash Number	Component Serial No.
Engine Assembly	8247-475200	11(-13)[e]	803
1. Thrust Chamber Assy	8247-470001	3	4
Fuel Valve Assy	8096-472105	5	2
(Fuel Valve)	8096-472110	3	2
(POSV)	8247-472015	3	20
Oxidizer Valve	8081-472050	5	2
Ox. Manifold Press. Sw.	8096-472065	19	A10594
2. Turbine Pump Assy	8247-474004	7	3
Fuel, GGSV	8247-472020	7	111 (115)[c]
Oxidizer, GGSV	8247-472025	11	121
Fuel, Dual Check	8247-472040	5	24
Ox., Dual Check	8247-472035	3	23
Fuel, Fill Valve	8247-472070	1	508112
Ox., Fill Valve	8247-472065	1	403202
Fuel, Bleed Valve	8247-472080	1	T2
Ox., Bleed Valve	8247-472075	1	7
Fuel Venturi Assy	8247-472060	1	T4
Ox. Venturi Assy	8247-472055	1	211
Fuel Tank Assy	8247-471202	1	003
Ox. Tank Assy	8247-471201	1	004
3. Electronic Gate	8247-472283	5	18
4. Relay Box	8247-472483	1	3 (10)[b]
Junction Box	8247-472383	7	001 [e]
5. Cable Assy	8247-475491	1	R12[a]
Cable Assy	8247-475481	1	3 (6)[b]
6. Ox. Feed Press. Sw.	8096-472065	17	A10573 (A10571)[d]
7. Duct Assy	8096-475105	11	

Legend:

a Acceptance test conducted with -491 harness
b Replaced cable assy and relay box due to short that occurred during checkout (1-16-66)
c Fuel GGSV S/N 111 leaked 9.35 cc/hr (max allowable 6.0 cc/hr) (replaced 1-16-66)
d Threads stripped in pressure switch case (replaced 2-9-66)
e Configuration change – retrofit with junction box – created -13 engine (2-8-66)

Table 6-7 (Cont.)

General Notes:

1. OFPS sensing line was not flight configuration due to instrumentation requirements.
2. Test harness shunt packs were installed for fuel and oxidizer GGSV electrical monitoring.
3. Test harness added to monitor OMPS and OFPS actuation during firing.

Table 6-8

ENGINE TEST INSTRUMENTATION AND DATA ACQUISITION

(Test Firings 2DT 281-296)

Parameter	Transducer Range	Transducer Monitor Point	Data Recording			
			Beckman 210	Strip Chart	Oscillograph	FM Tape
FSP-1	100 psia	Fuel pump inlet line	X	X	X	
FSP-2	100 psia	Fuel pump inlet line	X			
OSP-1	100 psia	Oxidizer pump inlet line	X	X	X	
OSP-2	100 psia	Oxidizer pump inlet line	X			
OVIP-1	2000 psia	BP-2	X	X	X	
OVIP-2	2000 psia	BP-2	X			
FVIP-1	2000 psia	BP-1	X	X	X	
FVIP-2	2000 psia	BP-1	X			
MVIP-1	1000 psig	Fuel pump motor venturi inlet	X			
MVIP-2	1000 psig	Fuel pump motor venturi inlet	X			
FVAP	1000 psig	BP-3	X		X	
FTP	1000 psia	Fuel tank ullage	X			
OTP	100 psia	Oxidizer tank ullage	X			
TMP-1	500 psig	DP-3	X	X	X	
TMP-2	500 psig	DP-3	X			
TCP-1	1000 psig	TC pressure sensing port	X			
TCP-2	1000 psig	TC pressure sensing port	X			
TCP-3	500 psig	TC pressure sensing port	X	X		
TCP-4	500 psig	TC pressure sensing port	X		X	
TCP-5	50 psia	TC pressure sensing port	X		X	
OFP	1000 psig	OFPS sensing line	X		X	
OMP-1	1000 psig	OMPS sensing line	X		X	
OMP-2	50 psia	OMPS sensing line	X		X	

Table 6-8 (Cont.)

Parameter	Transducer Range	Transducer Monitor Point	Data Recording			
			Beckman 210	Strip Chart	Oscillograph	FM Tape
N-Integ.				X		
N-Direct			X		X	
N-4					X	
LSP	50 psia	Lip seal pressurization line	X			
W_F-1 (total)		Fuel pump inlet	X		X	
W_F-2 (total)		Fuel pump inlet	X			
W_O-1 (total)		Oxidizer pump inlet line	X		X	
W_O-2 (total)		Oxidizer pump inlet line	X			
ODP	1000 psig	DP-8	X		X	
FDP	1000 psig	DP-9	X		X	
OCP	2000 psig	DP-1	X			
FCP	2000 psig	DP-13	X			
FST-1	See Note 1	Fuel pump inlet probe	X			
FST-2		Fuel pump inlet probe	X			
OST-1		Oxidizer pump inlet probe	X			
OST-2		Oxidizer pump inlet probe	X			
FSTT		Fuel start tank skin	X			
OSTT		Oxidizer start tank skin	X			
LST		Lip seal pressure line probe	X			
TCNT		TC nozzle skin	X			
TCIT		TC injector skin	X			
FTT		Fuel tank probe	X			
OTT		Oxidizer tank probe	X			
GGT		GG chamber skin	X			
TMT		Turbine manifold skin	X			
GBT		Gear case skin	X			
FSVT		Fuel solenoid valve skin	X			
OSVT		Ox. solenoid valve skin	X			

Table 6-8 (Cont.)

Parameter	Transducer Range	Transducer Monitor Point	Data Recording			
			Beckman 210	Strip Chart	Oscillograph	FM Tape
OFPST	See Note 1	OFPS skin	X			
FVT		Fuel valve skin	X			
OMPST		OMPS skin	X			
PSRBT		Relay box skin	X			
EGT		Electronic gate skin	X			
TCP	±50 psi	TC pressure sensing port				X
OMP	±50 psi	OMPS sensing line				X
Accel. Thrust	–	See Table 6-10				X
E_{tot}-1		Reference Figs. 6-6 and 6-7	X		X	
E_{tot}-2			X			
I_{tot}-1			X		X	
I_{tot}-2			X			
E-OSV					X	
E-FSV					X	
I-OSV					X	
I-FSV					X	
Fire			X		X	X
TCPS					X	
GGMPS					X	
Shutdown					X	
OSPS					X	

Table 6-8 (Cont.)

Parameter	Transducer Range	Transducer Monitor Point	Data Recording			
			Beckman 210	Strip Chart	Oscillograph	FM Tape
Engine Telltale (Electronic Gate)		Reference Figs. 6-6 and 6-7			X	
Relay Box Telltale					X	
Malfunction					X	

Note:

1. All temperature measurements taken with Chromel Aluminal thermocouples, type CA-150, with recording system calibration and range for the individual range of temperatures expected.

Table 6-9

SUMMARY OF ENGINE SEA LEVEL FWD TESTS

Test	Test Period (1966)	No. of Tests	Duration (sec)	Temperatures (° F)	Results
Fixed Thrust Calibration	Jan 16 – Jan 22	17	2, 30 & 90	45 to 75	Engine transient and steady state performance and operation satisfactory and within predicted limits
High & Low Temperature	Jan 29 – Feb 5	12	2, 15 & 30	0, 15 & 32 (Low) 100 & 140 (High)	Engine transient and steady state operation and performance verified over specification temperature range
Safety Limits	Feb 8	4	20	50 to 75	Engine operation and sequencing within prescribed limits at minimum and maximum voltage levels
Malfunction	Feb 8 – Feb 14	9	1, 3 & 20	50 to 60	Sea level malfunction modes of modified engine verified to be the same as the original -13 engine, i.e., safe and reliable shutdown obtained under the single malfunction conditions tested.
Dynamic	Jan 19 – Mar 2				Modified engine components installation and structural design verified as satisfactory under maximum specification vibration and shock conditions.
Procedures	Jan 5 – Feb 17	–	–	–	All procedural changes checked out and validated for use at AEDC, LMSC & ETR.

Table 6-10

ENGINE DYNAMIC MEASUREMENTS

Item Number	Parameter	Location and Orientation	Type Mounting	Type Transducer	Frequency Response	Accuracy	Range
A1	Engine Vibration	Thrust Chamber Lug – Longitudinal	Epoxy Bonded Mounting Block	Endevco 2242 C With Insulated Stud	5 to 4000 CPS	±5%	±150 g
A2		Thrust Chamber Lug – Yaw					±200 g
A3		Thrust Chamber Lug – Pitch					±200 g
A4		Gimbal Ring Assembly, Top – Longitudinal					±100 g
A5		Gimbal Ring Assembly, Bottom – Longitudinal					±200 g
A6		Thrust Mount, Gimbal Ring Attachment, Right Side – Longitudinal					±100 g
A7		Thrust Mount, Gimbal Ring Attachment, Left Side – Longitudinal					±200 g
A8		Thrust Mount, at Yaw Actuator Fitting, Longitudinal					±200 g
A9		Thrust Mount, at Pitch Actuator Fitting, Longitudinal					±300 g

Table 6-10 (Cont.)

Item Number	Parameter	Location and Orientation	Type Mounting	Type Transducer	Frequency Response	Accuracy	Range
A10	Engine Vibration	Electronic Gate – Longitudinal	Epoxy Bonded Mounting Block	Endevco 2242 C With Insulated Stud	5 to 4000 CPS	±5%	±100 g
A11	↓	Pressure Switch Relay – Longitudinal	↓	↓	↓	↓	±100 g
A12	↓	Thrust Chamber Lug – Longitudinal	Threaded Mounting Block	↓	↓	↓	±100 g
A13	↓	Gas Generator, Parallel to Geometric Axis of Generator Near Injector Head	Epoxy Bonded Mounting Block	↓	↓	↓	±400 g
A14	↓	Engine Test Rig – Longitudinal	↓	↓	↓	↓	±50 g
A15	↓	Engine Test Rig – Pitch	↓	↓	↓	↓	±50 g
16, LC-1	Engine Thrust	At Thrust Mount Attachment Point Upper Left	↓	Thrust Washer (Kistler)	↓	↓	0 to 7000 lbs.
17 LC-2	↓	At Thrust Mount Attachment Point Lower Left	↓	↓	↓	↓	↓

Table 6-10 (Cont.)

Item Number	Parameter	Location and Orientation	Type Mounting	Type Transducer	Frequency Response	Accuracy	Range
18, LC-3	Engine Thrust	At Thrust Mount Attachment Point Lower Right	Epoxy Bonded Mounting Block	Thrust Washer (Kistler)	5 to 4000 CPS	±5%	0 to 7000 lbs.
19, LC-4	↓	At Thrust Mount Attachment Point Upper Right	↓	↓	↓	↓	↓
20	Chamber Pressure	↓	↓	Kistler Transducer	↓	↓	↓

Table 6-11

DYNAMIC DATA FOR SEA LEVEL FWD TESTS

BAC Test No.	281	283	284	285	286	287	288	289	290	291	292	293	294	295	296	297	298
Temp. (°F) TMT/Prop.	40/46	46/53	604/74	526/76	511/76	459/78	456/71	84/56	96/48	215/63	329/60	367/61	96/47	97/47	212/60	33/41	35/45
Run Duration (sec)	90	31	2	2	2	2	2	2	2	2	2	2	2	2	2	2	91
Accelerometer Number	g's Vector (zero to peak)																
A-1	44	50	23	24	27	25	22	20	16	19	24	19	24	19	31	73	108
A-2	54	71	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
A-3	60	102	73	66	70	77	78	70	44	46	60	43	46	50	49	51	95
A-4	32	47	49	54	53	43	38	46	37	59	48	39	40	39	52	39	49
A-5	86	81	79	81	89	77	69	70	77	92	85	72	65	68	80	72	83
A-6	44	37	35	47	25	41	44	38	40	40	44	36	41	44	41	41	38
A-7	63	64	36	41	43	40	49	59	78	48	47	67	54	50	53	71	72
A-8	66	71	60	61	67	80	65	78	55	71	76	72	53	58	61	53	62
A-9	84	71	68	63	68	74	77	73	74	82	77	78	87	67	94	68	66
A-10(1)	–	–	17	15	14	9	8	14	12	16	15	18	12	12	18	17	10
A-11	16	6	10	6	16	12	17	10	7	15	10	11	10	7	9	14(1)	9(1)
A-12	45	66	72	64	76	78	53	56	51	62	75	69	55	58	76	42	64
A-13	71	93	120	53	–	–	–	–	–	–	–	–	–	–	–	79	67(1)
A-14	5	6	4	6	4	4	2	2	6	10	10	10	8	13	8	3(1)	4
A-15	6	8	5	8	7	3	7	2	5	10	8	6	8	8	8	16(1)	4
Ave. g's(2)	55	63	54	48	50	51	47	49	45	50	51	48	44	43	51	52	60
Ave. g's(3)	53	59	48	50	51	52	50	52	38	53	53	49	47	45	53	54	65
Load Cells	Pounds Force																
LC-1	–	3100	2425	2340	2115	2078	1968	1892	2195	2495	2240	2280	2410	2152	2195	2200	2095
LC-2	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–	2970	2886
LC-3	–	3260	3260	3220	2585	2681	2585	2325	2980	3105	3182	2980	2900	2898	2980	2940	2940
LC-4	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–	2855	–
Total Force lbs.	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–	10965	–

(1) 60 cps noise on data, however the data is considered valid
(2) Does not include accelerometers Nos. 14 and 15 (test rig accelerometers)
(3) Includes accelerometers A-1 through A-10 for comparison with similar data from AEDC tests
– No data due to instrumentation discrepancies or low acceleration levels

Table 6-11 (Cont.)

BAC Test No.	299	300	301	302	303	304	305	306	307	308	309	310	311	312	313	314
Temp.(°F)TMT/Prop.	2/0	519/13	468/16	18/32	484/35	419/35	40/133	550/143	511/143	96/103	603/105	507/107	50/54	363/62	309/73	421/71
Run Duration (sec)	30	2	2	30	2	2	15	2	2	31	2	2	20	20	21	20
Accelerometer Number	g's Vector (Zero to Peak)															
A-1	68	48	44	34	45	50	83	41	43	52	36	48	40	39	42	39
A-2	55	75	53	66	74	56	67	37	–	47	27	40	41	37	64	47
A-3	29	29	33	51	34	40	41	27	27	33	23	23	25	27	42	32
A-4	10	29	36	57	47	36	6	8	6	42	71	68	39	40	48	41
A-5	88	65	75	78	66	72	118	90	95	83	94	91	61	71	83	68
A-6	48	49	49	30	37	36	86	39	39	–	22	22	38	26	28	27
A-7	62	49	47	52	35	41	72	40	35	39	39	41	38	40	36	35
A-8	47	52	48	54	47	41	72	44	37	24	24	29	47	45	48	35
A-9	78	77	90	97	83	89	97	65	98	76	64	76	95	60	72	58
A-10(1)	14	21	25	14	20	12	–	–	–	17	18	6	–	–	–	–
A-11	9	17	11	15	10	8	53	20	13	11	15	10	12(1)	14(1)	11	11
A-12	42	44	36	53	46	41	42	56	58	64	84	43	47	36	38	45
A-13	88	100	113	84	–	–	104(1)	–	–	168	–	–	116	130	114	124
A-14	3	9	8	8	5	6	24	5	7	6	5	3(1)	3	6	3	6
A-15	6	9	6	–	7	4	43	6	8	9	5	9(1)	8	6	8	7
Ave. g's(2)	49	50	51	53	45	44	70	43	45	55	43	41	50	47	52	47
Ave. g's(3)	50	50	50	53	49	47	71	43	48	46	42	44	47	43	51	42
Load Cells	Pounds Force															
LC-1	2260	2345	2490	2455	2610	2455	3695	2820	3030				3292	3145	3056	3142
LC-2	3001	3145	3190	3150	3150	3150		2660	2345				2360	2510	2040	2474
LC-3	3002	3050	2825	2970	2878	2845	4150	4440	4420				4640	4650	4540	–
LC-4	2890	3230	3180	3030	3115	3200	2442	2290	2105				2075	1989	1933	2392
Total Force lbs.	11153	11770	11685	11605	11753	11650	–	12210	11900				12367	12294	11569	–

(1) 60 cps noise on data, however the data is considered valid.
(2) Does not include accelerometers Nos. 14 and 15 (test rig accelerometers).
(3) Includes accelerometers A-1 through A-10 for comparison with similar data from AEDC tests.
– No data due to instrumentation discrepancies or low acceleration levels.

Table 6-11 (Cont.)

BAC Test No.	315	316	317	318	319	320	321	322	323	324
Temp. (° F) TMT/Prop.	84/57	43/49	136/64	183/60	183/60	237/58	237/58	278/58	278/58	254/60
Run Duration (sec)	3	30	1	1	0.5	1	0.1	1	0.4	20
Accelerometer Number	g's Vector (Zero to Peak)									
A-1	31	33	37	37	60	37	40	49	55	48
A-2	36	23	30	40	80	50	50	43	65	55
A-3	19	27	29	27	54	29	29	34	44	54
A-4	22	83	60	46	76	57	62	71	48	61
A-5	57	90	71	71	125	90	61	76	78	100
A-6	26	–	–	–	–	–	–	–	–	–
A-7	40	45	53	48	38	50	45	50	48	53
A-8	51	100	58	53	75	55	65	55	50	55
A-9	60	106	78	53	71	71	67	78	53	78
A-10[1]	–	–	–	–	–	–	–	–	–	–
A-11	13	4	4	3	5	5	3	3	3	3
A-12	34	55	56	44	77	44	35	48	47	77
A-13	114	122	146	156	205	171	200	225	239	195
A-14	3	7[1]	13[1]	9[1]	7[1]	7[1]	5[1]	8[1]	4[1]	5[1]
A-15	7	2[1]	4[1]	50[1]	3[1]	6[1]	3[1]	6[1]	9[1]	6[1]
Ave. g's[2]	42	63	57	53	79	60	60	67	66	71
Ave. g's[3]	38	63	52	47	72	55	52	57	55	63
Load Cells	Pounds Force									
LC-1	–	4240	3680	2405	2860	2830	2572	3450	2630	3280
LC-2	–	3200	2412	1280	1485	2008	1310	1804	1165	2010
LC-3	–		–	4110	3830	–	–	–	4240	4240
LC-4	–	2590	2305	1614	1728	2160	1902	2075	2020	2275
Total Force lbs.			–	9409	9903	–	–	–	10055	11805

(1) 60 cps noise on data, however the data is considered valid.
(2) Does not include accelerometers Nos. 14 and 15 (test rig accelerometers).
(3) Includes accelerometers A-1 through A-10 for comparison with similar data from AEDC tests.
– No data due to instrumentation discrepancies or low acceleration levels.

Table 6-12

POWER SPECTRAL DENSITY DATA FOR SEA LEVEL FWD TESTS

Accelerometer Numbers and g (RMS) Levels

Test	A1	A2	A3	A4	A5	A6	A7	A8	A9	A10	A11	A12	A13	A14	A15	Aver-age
287	3	–	4	6	14	6	5	7	7	12	7	14	3	4	3	7
297	6	–	3	4	9	5	5	6	5	15	8	11	11	5	4	7
299	5	3	3	2	7	–	4	5	6	5	4	3	13	2	1	5
301	4	4	17	2	1	–	1	1	0.5	3	7	2	3	5	9	4
306	7	5	5	2	12	10	7	10	8	8	5	5	4	2	1	6
307	8	5	6	1	13	8	7	10	8	10	4	3	5	2	1	6
308	7	3	3	4	9	2	6	8	6	8	4	4	11	1	1	5
310	6	5	5	6	13	4	7	10	7	7	6	17	4	2	2	7
324	5	3	3	6	5	13	4	5	5	9	4	12	17	2	2	6

–No data for these accelerometers.